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A NUCLEAR-ELECTRIC PROPULSION SYSTEM

BY R. W. BUSSARD, B.S.

The use of electric fields to produce thrust by the acceleration of ions or charged particles has been proposed many times.[1,2,3,4] In considering such a scheme attention should be given to the possibility of system optimization by proper choice of particle mass, charge, and accelerating voltage. In particular the system performance criteria should be related to the limitations imposed by space-charge within the accelerator.

It is the purpose of this paper to present an analysis of the potentialities of particle accelerators as thrust producers and to correlate their performance with the other components of a complete self-contained system as required for free-space operation. The mks system of units will be used throughout.

Particles and Accelerator

Consider an arbitrary particle of mass M proton masses with a charge of Q electronic charges. It is easy to show[5] that the space-charge limited current density within a time-invariant accelerating field of E_b volts (in parallel plate geometry) is given by

$$j = \frac{\epsilon_o}{2{\cdot}25} \sqrt{2(e/m_p)\,(Q/M)\,E_b}\,(E_b/d^2)\ \text{amp/m}^2 \qquad (1)$$

where ϵ_o is the permittivity of free space, e is the electronic charge in coulombs, m_p is the mass of one proton in kg., and d is the length of path within the uniform accelerating field in m. The thrust due to this accelerator is proportional to the momentum change imparted to the charged particle stream and is given by

$$T = wv_e/g_c\ \text{kg}. \qquad (2)$$

for initial particle velocity small compared to final velocity. Here w is particle weight flow rate in kg./sec., v_e is accelerator exit velocity in m./sec., and g_c is the gravitational acceleration, 9·80 m./sec^2.

The particle velocity at exit from the accelerating field is given by

$$v_e = \sqrt{2(e/m_p)\,(Q/M)\,E_b}\ \text{m./sec}. \qquad (3)$$

Electric current is simply the flow of electric charge per unit time and since

each charge is associated with a mass the current is directly related to the particle weight flow rate. One ampere is defined as a flow of $6{\cdot}28 \times 10^{18}$ electronic charges per second, thus the weight flow rate is related to the current by

$$w = 6{\cdot}28 \times 10^{18}(Mm_p/Q)i \text{ kg./sec.} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (4)$$

where i is the total current in amps., and is given by

$$i = j\,S^2 \text{ amps.} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (5)$$

where S^2 is the cross-sectional area of the accelerating field, in m^2.

By combination of equations (1) through (5) the thrust per unit cross-section of accelerator tube (or accelerating field) can be shown to be

$$T/S^2 = 5{\cdot}58 \times 10^{18}(e\epsilon_o/g_c)(E_b/d)^2 \text{ kg./m.}^2 \quad .. \quad .. \quad .. \quad .. \quad (6)$$

It is noted that the accelerator thrust per unit area as limited by space-charge is completely independent of the particle mass, charge, or charge-to-mass ratio and depends only upon the gradient of the accelerating field. Thus from this standpoint (neglecting relativistic effects) electrons are as satisfactory as "propellants" as are mercury ions or charged iron filings. Although the space-charge limited current is a direct function of the charge to mass ratio (see equation (1)) the heavier charged particles have a lower mobility and hence a longer residence time in the region of effective space-charge, thus nullifying the effect of greater mass flow per unit current.

However, the particle charge-to-mass ratio is important in determining the power consumption of an accelerator thrust system. The total electrical power is iE_b and is found from equations (1) and (5) as

$$P_e = (10^{-6}\epsilon_o/2{\cdot}25)(E_b/d)^2(S)^2\sqrt{2(e/m_p)\,(Q/M)\,E_b} \text{ MW.} \quad .. \quad .. \quad (7)$$

The power consumption per unit thrust is of more interest. This is determined from equations (6) and (7) to be

$$P_e/T = 1{\cdot}13 \times 10^{-25}\,g_c\sqrt{(Q/M)\,(E_b/em_p)} \text{ MW./kg.} \quad .. \quad .. \quad (8)$$

The physical constants appropriate to the mks system of units are

$$\epsilon_o = 10^{-9}/36\pi$$
$$g_c = 9{\cdot}80 \text{ m./sec.}^2$$
$$e = 1{\cdot}60 \times 10^{-19} \text{ coulombs}$$
$$m_p = 1{\cdot}66 \times 10^{-27} \text{ kg.}$$

Application of these values to equations (6) and (8) yield

$$T/S^2 = 8{\cdot}06 \times 10^{-13}\,(E_b/d)^2 \text{ kg./m.}^2 \quad .. \quad .. \quad .. \quad .. \quad (9)$$

and

$$P_e/T = 6{\cdot}79 \times 10^{-2}\,(Q/M)^{1/2}\,E_b{}^{1/2} \text{ MW./kg.} \quad .. \quad .. \quad .. \quad (10)$$

It is obvious, from equation (9), that voltage gradients of the order of 10^6 to 10^7 volts/m. are required if noticeable thrust is to be produced from accelerators of reasonable size.

For estimation of system flight performance it is desirable to know the

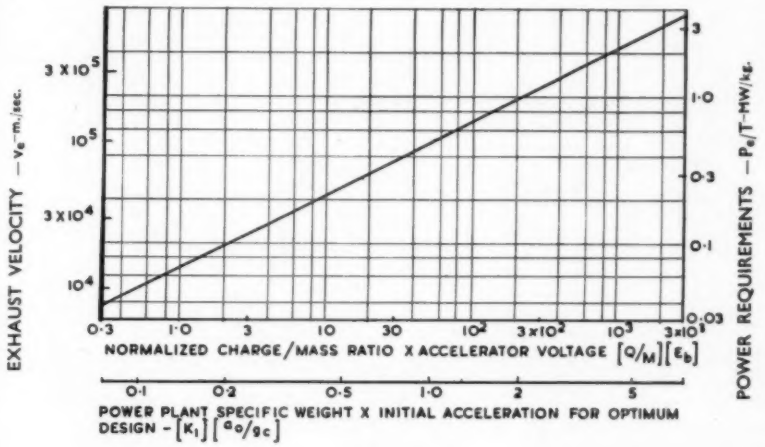

FIG. 1

effective particle "exhaust velocity," as given by equation (3) reduced by use of the above physical constants. Thus

$$v_e = 1{\cdot}39 \times 10^4 \, (Q/M)^{1/2} \, E_b^{1/2} \text{ m./sec.} \quad .. \quad .. \quad .. \quad .. \quad (11)$$

Figure 1 shows the variation of P_e/T and v_e with the product (Q/M) (E_b).

System and Vehicle

In order to gain an understanding of the effect of variation of the power system and accelerator characteristics on overall vehicle performance, it is of interest to determine the relations between vehicle weight, thrust, and gross specific impulse, and particle mass, charge, and accelerating voltage.

Assume a vehicle and propulsion system as outlined in Figure 2. Here the propulsion system weight is the sum of the propellant weight (m_a), tankage weight (m_t), accelerator weight (m_a), thermal energy source weight (m_s), electrical generating equipment weight (m_e), and waste heat dump weight (m_r). The vehicle gross weight (m_o) is the sum of the propulsion system weight and "dead load" weight (m_d). The dead load weight is defined to include crew compartment, cargo or payload, any guidance or other equipment, and any vehicle structure not included in the propulsion system weight.

Tankage weight is assumed to be a fraction K_2 of the weight of propellant carried, thus

$$m_t = K_2 m_a \text{ kg.} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (12)$$

The weight of accelerator, energy source, electrical generating equipment, and heat dumps is assumed proportional to the electrical power requirements, as

$$m_a + m_s + m_e + m_r = K_1 P_e \text{ kg.} \quad .. \quad .. \quad .. \quad .. \quad (13)$$

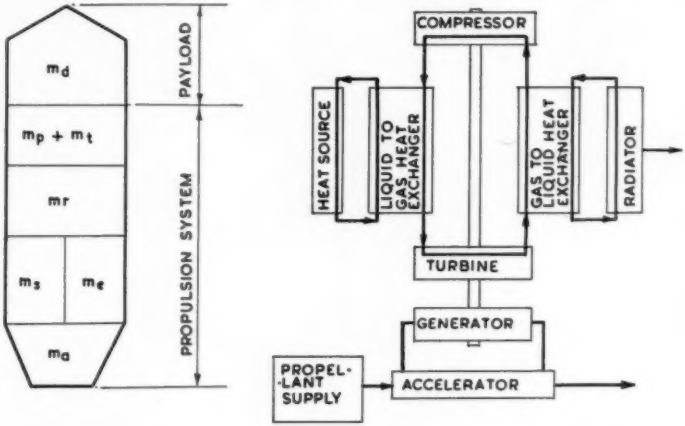

FIG. 2

where the overall specific weight, K_1, is the sum of the individual specific weights of each component of the power plant.

Combining equations (12) and (13) with the definition of gross weight the total vehicle weight is given by

$$m_o = m_d + m_q(1 + K_2) + K_1 P_e \text{ kg.} \quad .. \quad .. \quad .. \quad .. \quad (14)$$

Since the total impulse imparted to the vehicle is the product of the thrust (T) and the operating time (t_b), assuming constant thrust, the overall gross specific impulse of the complete vehicle is given by

$$I_{oa} = Tt_b/m_o \text{ sec.} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (15)$$

By use of equations (2) and (14) the overall specific impulse can be expressed as

$$I_{oa} = \frac{m_q v_e/g_c}{m_d + m_q(1 + K_2) + K_1 P_e} \text{ sec.} \quad .. \quad .. \quad .. \quad .. \quad (16)$$

The electrical power output can be obtained from the rate of kinetic energy addition to the propellant, and is

$$P_e = 4{\cdot}90 \times 10^{-6}\,(m_q/t_b)\,(v_e^2/g_c) \text{ MW.} \quad .. \quad .. \quad .. \quad .. \quad (17)$$

By use of the relation $(m_d/m_q) = (m_d/m_o)\,(m_o/m_q)$ and the familiar mass-ratio equation $m_o/(m_o - m_q) = e^{\xi}$ (where $\xi = \Delta v/v_e$) the dead load weight can be expressed by

$$m_d = m_q(m_d/m_o)\,\phi_\xi \text{ kg. where } \phi_\xi = e^{\xi}(e^{\xi} - 1) \quad .. \quad .. \quad .. \quad (18)$$

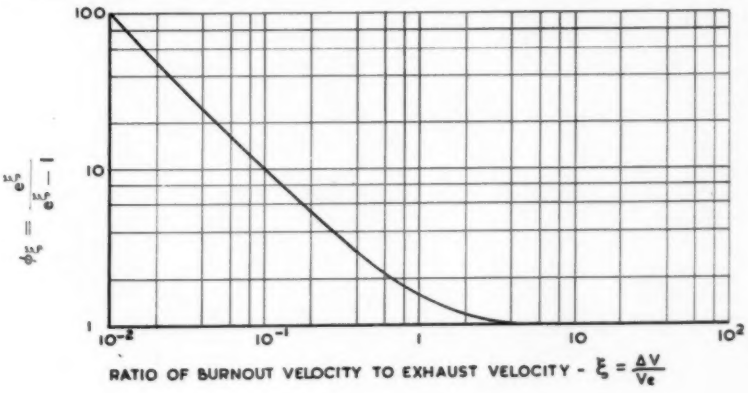

FIG. 3

For reference the function ϕ_ξ is shown graphically *vs* $\Delta v/v_e$ in Figure 3.

Combining equations (17) and (18) with equation (16) the overall specific impulse is given in terms of vehicle performance factors, particle exhaust velocity, and accelerator operating time, as

$$I_{oa} = [(\lambda g_c/v_e) + (4{\cdot}90 \times 10^{-6} K_1 v_e/t_b)]^{-1} \text{ sec.} \quad .. \quad .. \quad .. \quad (19)$$

where $\lambda = 1 + K_2 + (m_d/m_o)\phi_\xi$. It is easy to show that the overall specific impulse reaches a maximum at a value of v_e given by

$$v_e^2 = 2{\cdot}03 \times 10^5 (\lambda g_c/K_1) t_b \text{ (m./sec.)}^2 \quad .. \quad .. \quad .. \quad .. \quad (20)$$

The maximum overall specific impulse is then

$$I_{moa} = 225\sqrt{t_b/\lambda g_c K_1} \text{ sec.} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (21)$$

Note that this optimum specific impulse depends only upon the system operating time, for any given vehicle performance requirements and structural specific weight capabilities.

As a matter of reference it should be noted that the conventional propellant specific impulse is given by

$$I_{sp} = 2\lambda I_{moa} \text{ sec.} \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad .. \quad (22)$$

This value for optimum exhaust velocity, given by equation (20), coupled with equation (11) yields a relation between particle charge-to-mass ratio, accelerating voltage, and system operating time as

$$(Q/M)(E_b) = 1{\cdot}05 \times 10^{-3} (\lambda g_c/K_1) t_b = 2{\cdot}07 \times 10^{-8} (\lambda g_c I_{moa})^2$$
$$= 5{\cdot}17 \times 10^{-9} (g_c I_{sp})^2 \quad .. \quad (23)$$

The thrust-to-weight ratio is of interest since it will serve as a measure of the applicability of a particle accelerator thrust system for use within a gravitational

force field. This is equal to the vehicle acceleration in free fall and is given by

$$T/m_o = a_o/g_c = I_{moa}/t_b = 225/\sqrt{\lambda g_c K_1 t_b} = 5{\cdot}07 \times 10^4/\lambda g_c K_1 I_{moa}$$
$$= 1{\cdot}014 \times 10^5/g_c K_1 I_{sp} \quad .. \quad (24)$$

The thrust-to-weight ratio will be large only for short operating times or for low values of maximum specific impulse. For a given set of vehicle parameters high specific impulse requires long operating time and low acceleration, hence such propulsive systems are clearly most useful in relatively field-free conditions in space.

Using equation (24), equations (10), (11) and (23) can be rewritten in much simpler form involving only the power plant specific weight parameter and the initial vehicle acceleration. These equations are presented here for ready reference.

$$(Q/M)\,(E_b) = 53{\cdot}3/(K_1 a_o/g_c)^2 \quad .. \quad .. \quad .. \quad .. \quad .. \quad (25)$$

$$P_e/T = 0{\cdot}494/(K_1 a_o/g_c) \text{ MW./kg.} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (26)$$

$$v_e = 1{\cdot}01 \times 10^5/(K_1 a_o/g_c) \text{ m./sec.} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (27)$$

From the point of view of the vehicle designer it is evidently very desirable to reduce the weight of the electrical plant as much as possible. Accordingly, it is pertinent to examine briefly the weight characteristics of each of the components concerned.

Since heat rejection in space can only be practically accomplished by thermal radiation the radiator weight is of importance. For operation at 600° C, with an emissivity of 1·0, roughly 300 ft.^{2} of radiating surface are required per thermal MW. radiated. Assuming a weight of 2 kg./ft.,^{2} including working fluid, the radiator specific weight is given by

$$m_r/P_e = 600\,(1 - \eta_{oa})/\eta_{oa} \text{ kg./electrical MW.} \quad .. \quad .. \quad .. \quad (28)$$

where η_{oa} is the overall thermal-electrical conversion efficiency of the power plant. Of course, operation at higher temperature would reduce the area required and therefore the weight per unit power; however, the overall cycle efficiency when using a temperature limited heat source will go down as the sink temperature is raised. The determination of optimum sink temperature is beyond the scope of this paper, particularly since thorough analysis of the entire radiation heat sink problem has been previously presented.[6]

The size and hence weight of a fission reactor heat source is limited only by the heat transfer capabilities of the fissioning volume (reactor core). Heat flux densities up to 1 thermal MW./ft.^{3} are easily attainable with liquid metal coolants and moderate temperature driving potentials.[7] Heat generation rates higher than this value have been achieved with the M.T.R. (Materials Testing Reactor) at Arco, Idaho.[8] Assuming a mean reactor density of about 100 kg./ft.^{3} of core volume, typical of large power reactors such as the X-10 graphite pile,[9] the reactor weight is then given crudely by

$$m_{s_1}/P_e = 100/\eta_{oa} \text{ kg./electrical MW.} \quad .. \quad .. \quad .. \quad .. \quad (29)$$

This figure neglects the fact that a minimum reactor size will be required for nuclear criticality, regardless of power level. However, for use of the above

specific weight this consideration becomes of second order importance for power outputs greater than about 20–40 thermal MW.

For various reasons related to the operation of the heat engine it may be desirable to isolate the reactor from the rest of the power plant circuitry. To do so requires that a secondary heat exchanger be provided to transfer heat from the reactor coolant to an independent heat engine working fluid. Gas-to-gas exchangers capable of transferring up to 2 kW./kg. have been developed for many years.[10] It is evident that this performance could be doubled in a liquid-to-gas heat exchanger in which the liquid side resistance to heat transfer is very low compared to that on the gas side. Heat transfer performance greater than this has been achieved in liquid metal to air exchangers.[11] Thus for heat transfer to gas as a heat engine working fluid the secondary heat exchanger weight is taken to be

$$m_{s_1}/P_e = 250/\eta_{oa} \text{ kg./electrical MW.} \quad .. \quad .. \quad .. \quad .. \quad (30)$$

Available high performance conventional electrical generators[12] have a specific weight of about

$$m_{e_1}/P_e = 1{,}000 \text{ kg./electrical MW.} \quad .. \quad .. \quad .. \quad .. \quad (31)$$

As recently suggested[13] it may be possible to produce electrical power much more cheaply than this, weight-wise, by use of electro-static generators. The performance figures for such devices must await their development hence cannot be included here.

For a pressurized closed gas cycle heat engine system, using inert gas such as He or N_2 as the working fluid, it should be possible to build a high performance turbine-compressor with a specific weight of the order of

$$m_{e_2}/P_e = 500 \text{ kg./electrical MW.} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (32)$$

Such a system would be similar to the Ackeret-Keller regenerative units[14] in use for some years. The desirability of regeneration can only be determined by a weight analysis made balancing radiator weight against regenerator weight. At first glance the regenerator seems an inefficient (weight-wise) device, being a gas-to-gas heat exchanger. It appears doubtful that increased cycle efficiency would be sufficient to pay its freight in space.

The lightest of the propulsion system components should be the particle accelerator, basically consisting of a charged particle source and an accelerating grid structure supported on a light-weight framework. Although admittedly a guess, accelerator weight could be as low as

$$m_a/P_e = 100 \text{ kg./electrical MW} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (33)$$

Assuming this value and an overall efficiency of 25 per cent. the specific weight of the complete propulsion system, less propellant and tankage, will be

$$K_1 = 4{,}800 \text{ kg./electrical MW.} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (34)$$

for use of conservative design and standard or near-standard equipment. At best a factor of 10 improvement might be hoped for from refined development of individual components.

As an example of the possibilities with even the rather conservative approach used above, consider a vehicle in which it is desired that the propellant specific impulse be 10,000 sec., 30 times the best attainable by chemical rockets to-day.

For $K_1 = 4{,}800$ kg./electrical MW. equation (24) can be used to show that $a_o = 2{\cdot}17 \times 10^{-4} g_c$. Though small this acceleration is not trivial. The gravitational acceleration of the Sun's field at the orbital distance of the Earth is only $6{\cdot}04 \times 10^{-4} g_c$! Specifying an overall specific impulse of 3,000 sec. the thrust unit operating time is found to be $1{\cdot}38 \times 10^7$ sec., or about 5·3 months. The value of λ is determined from equation (22) to be $\lambda = 1{\cdot}667$. Assuming $m_o = 10^5$ kg., then $T = 21{\cdot}7$ kg. Now for $(Q/M)\ (E_b) = 50$ from equation (23) or equation (25), equation (10) or equation (26) shows that $P_e/T = 0{\cdot}48$ MW/kg., so that $P_e = 10{\cdot}4$ electrical MW. Since $K_1 = 4{,}800$, the weight of equipment necessary to produce the electrical power is 5×10^4 kg. For an accelerator voltage gradient of $E_b/d = 10^7$ volts/m. equation (9) yields $T/S^2 = 80{\cdot}6$ kg./m.^{2} so that $S^2 = 0{\cdot}25$ m.^{2} This corresponds to a circular accelerator grid about 0·56 m. or 1·85 ft. in diameter. The propellant weight flow rate was computed from equation (2) and found to be $w = 2{\cdot}17 \times 10^{-3}$ kg./sec. The total propellant weight is thus 3×10^4 kg. Now for a tank 10 per cent. as massive as its contained propellant, $K_2 = 0{\cdot}1$, thus $(m_d/m_o)\ \phi_\xi = 0{\cdot}567$. For $K_2 = 0{\cdot}1$ the tank weight is $m_t = 3 \times 10^3$ kg. Summing up all the weights it is found by subtraction that $m_d = 1{\cdot}7 \times 10^4$ kg., thus $(m_d/m_o) = 0{\cdot}170$ and ϕ_ξ must be 3·33. Therefore $e^\xi = 1{\cdot}43$ and $\xi = \Delta v/v_e = 0{\cdot}358$. Since $v_e = 9{\cdot}80 \times 10^4$ m./sec., (by definition of $I_{sp} = 10{,}000$ sec.) then $\Delta v = 3{\cdot}50 \times 10^4$ m./sec. or 21·8 miles/sec. The total impulse over the entire motor operating time has the enormous value of 3×10^8 kg./sec. This is over 200 times the total impulse required by a V-2 rocket missile for normal flight. Note that for the same specific impulse values, a reduction in K_1 results in a corresponding reduction in system operating time and an increase in vehicle acceleration.

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LEGAL PROBLEMS OF UPPER SPACE

By Professor John Cobb Cooper*

To-day neither lawyers nor governments are prepared to state the legal flight rules applicable to presently operating rockets and planned satellites. For the second time in the present century science and engineers have far outstripped the law. Such delay may be normal where legal rules must depend on known and accepted factual situations, but the gap between technological and legal progress must never be permitted to become too wide.

The present problem confronting us is this: rockets, satellites and other spacecraft do not fall within existing national or international regulatory provisions, nor is there any agreement as to what, if any, national or international regulation is applicable to *space* above the atmosphere where such rockets and satellites will normally be used. This situation can lead to grave international misunderstanding if permitted to continue too long.

Flight technology first outdistanced the law in the early years of the present century when it became necessary to determine whether aircraft and balloons were vehicles like automobiles or whether they had nationality like ships; also, whether the atmospheric space in which they operated was or was not part of the territory of the subjacent State. The manner in which these questions were settled reveals the extent of the dangerous legal hiatus which now confronts us.

The following appears to be the present legal situation:

(*a*) Both the 1910 International Air Navigation Conference in Paris and the Chicago Convention of 1944 have dealt only with those flight instrumentalities which derive support in the atmosphere from reactions of the air, such as the balloon or airplane, and have not dealt with such instrumentalities as rockets, satellites, and other spacecraft which are designed to move through space without atmospheric support.

(*b*) The Chicago Convention contains no definition of "airspace" but it may well be argued that, as it was adapted from the Paris Convention, it deals with no areas of space other than those parts of the atmosphere where the gaseous air is sufficiently dense to support balloons and airplanes. The highest flight by any unmanned balloon up to the present time is 140,000 ft., by a manned balloon 72,395 ft., and the highest airplane flight is 90,000 ft.

(*c*) Nothing in the Chicago Convention precludes the possibility of State sovereignty being extended by International agreement, or by unilateral force, above the areas in which the airplane and balloon can be used, but there is certainly no basis on which any customary international law can as yet be considered applicable to such higher areas.

(*d*) Air space over the high seas is now free for use by all.

* *Extracts from an address delivered to the annual meeting of the American Society of International Law, April 26, 1956.*

In July, 1955, the United States announced that it would support a series of satellite flights as part of the wide scientific investigation of the 1957–1958 International Geophysical Year. Some of the details of the proposed flights have now been disclosed.

Two authorities have already dealt with the legal problems of this flight. Mr. Andrew G. Haley[1] presented a paper in November, 1955, in which he seemed to suggest that the areas of space above the atmosphere to be used by the satellite might be subject to some sovereign control of the subjacent States, but that failure of any State to object to the International Geophysical Year satellite programme at the time of its announcement was all that was required in order to make the completion of the programme possible. He added that "The scientists have benefited mankind as a whole in a field where the lawyers might well have failed."

Quite a different thesis has been put forward by Mr. C. Wilfred Jenks, an Associate of the Institute of International Law.[2] In his article Mr. Jenks noted the announcement by the White House of a satellite programme "to circle the earth in 90 minutes at a height of 300 miles." His legal position is, apparently, that space beyond the atmosphere of the earth is and must always be incapable of appropriation by the projection into such space of any particular sovereignty based on a fraction of the earth's surface. He argues that the acceptance of such complete international freedom in these areas of space is required by astronomical and physical facts, and he contends that only activities "within the atmosphere of the Earth would appear to be susceptible of the degree of control similar in general nature to that which can be exercised in territorial waters or over a wider maritime frontier belt." It would therefore appear that Mr. Jenks denies the existence or possibility of national sovereignty in areas of space beyond the atmosphere—say from 300 miles above the Earth's surface upward. He continues:

> "It would seem important to accept this principle fully from the earliest stages of the exploitation and exploration of space, and it is of interest that the United States plan for launching space satellites appears to be based upon it. There is no indication in the United States plan that it is proposed to negotiate passage agreements with the subjacent sovereignties. Moreover, such rights cannot be claimed under the International Civil Aviation Convention, since, even assuming the Convention to be applicable beyond the atmosphere and disregarding the fact that certain States, including the U.S.S.R., are not parties to it, the Convention provides that pilotless aircraft will not be flown over the territory of contracting States without special authorization."

The statements by Mr. Haley and Mr. Jenks point up the importance of the problem to be considered—namely, what is the legal status of the space beyond the atmosphere where rockets and satellites can be operated without undue atmospheric interference?

The only practical way to solve the questions as to the legal status of areas above those covered by a strict construction of Article I of the Chicago Convention will be the adoption of some form of international agreement.

Such a new convention might include these solutions:

(*a*) Reaffirm Article I of the Chicago Convention, giving the subjacent State full sovereignty in the areas of atmospheric space above it, up to the height where "aircraft" as now defined, may be operated, such areas to be designated "territorial space."

(*b*) Extend the sovereignty of the subjacent State upward to 300 miles above the Earth's surface, designating this second area as "contiguous space," and provide for a right of transit through this zone for all non-military flight instrumentalities when ascending or descending.

(*c*) Accept the principle that all space above "contiguous space" is free for the passage of all instrumentalities.

These solutions would aid future peaceful use of rockets and satellites and would seem to provide reasonable security for the subjacent State. At the same time, the territory of the State would be extended upward even beyond the areas in which it might make its normal laws effective. For I venture to suggest that, due in part to the physical problems involved, in part to the enormous speeds of the flight instrumentalities concerned, as well as many other difficulties, it will be most unlikely that any State can make its normal day-by-day laws effective very high in space. I do not deny the possibility that with modern weapons, such as guided missiles, a State may exercise *military command* quite high into space, provided it is certain that its activities are within the areas which are really above its own territory. But I must differentiate between such military control as may be involved in shooting down an intruder, and the normal civil control that a State must have day by day to enforce in its territory the peacetime laws under which men live together.

Another problem to be dealt with is the difficult question of nationality. The whole theory of nationality, as derived from the law of the sea, is based on the concept that when a State gives to a ship the right to use its flag, such State assumes certain international responsibilities for the good conduct of that ship on the high seas and in foreign ports and at the same time acts as the protector of the ship to enforce its international rights. Under the Chicago Convention, aircraft are given the same characteristics of nationality. In addition, Article VIII deals with aircraft "capable of being flown without a pilot," and it would seem that such pilotless aircraft also have nationality. While the application of the rule of nationality to rockets and satellites may be difficult, nevertheless if upper space is to be free like the high seas, then certainly a State must be prepared to be responsible for the international good conduct of its rockets and satellites; otherwise chaos might result. Nationality must be considered when these new types of flight instrumentalities are brought within the sphere of international regulation.

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EXTRA-TERRESTRIAL MINING

By A. J. W. Rozelaar,* B.Sc. (Hons.), A.M.I.M.M.

SUMMARY

This paper takes note of the reasons for extra-terrestrial mining and the materials likely to be required. It goes on to consider what minerals are, in fact, likely to be encountered.

It is emphasized from this that the raw materials for plastics, concrete, etc., will not be available, and that greater use will have to be made of metals to compensate for this.

Prospecting methods used on Earth are mentioned, and their applicability elsewhere is considered. A suggested prospecting procedure for extra-terrestrial use is given.

A table is given showing acid radicles found in nature, and the elements associated with them.

An alphabetical list of 48 elements and groups of elements is also given showing their most likely mode of occurrence.

Basic Problems of Extra-Terrestrial Mining

This paper considers the problems associated with mining operations upon the Moon or any airless body. Little space need be devoted to bodies having atmospheres of comparable density to that of Earth, since the problems there will be similar to those already dealt with every day down terrestrial mines.

The reasons for and advantages of extra-terrestrial mining are as follows:—

(1) Exhaustion of a particular raw material on Earth.

(2) The need for various raw materials upon the particular astronomical body in question.

(3) The use of excavations for living and/or storage accommodation.

Little amplification is necessary in connection with the first reason, except to point out that the high costs of interplanetary transportation make it unlikely that extra-terrestrial resources would be exploited for this reason unless all substitutes for the material in question had already been tried and found wanting, or when the material was extremely valuable.

The second reason requires rather more elaboration. It seems logical that if materials can in the long run be obtained more cheaply and easily by local mining than by importation from Earth, then all efforts should be made to operate mines on the spot. "In the long run" is a necessary qualification, since initially all supplies of machinery and materials must be brought from Earth; hence until the weight and/or value of the locally produced article exceeds that of the machinery and materials and their transport, the project will not be economic even by interplanetary standards.

The main types of materials likely to be sought after in extra-terrestrial situations for local use are as follows:—

(*a*) *Constructional materials of all kinds.* It should be remembered that materials useless on Earth may be valuable elsewhere. For example, two very light metals, lithium (specific gravity 0·53) and calcium (specific gravity 1·53), might be very much more useful where there is no atmosphere to corrode them.

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(*b*) *Materials for the maintenance of life.* These include materials containing oxygen, carbon, hydrogen, nitrogen, phosphorus, sodium, chlorine, etc. If it were possible to separate these elements and then where required recombine them for use, a local atmosphere could be created and basic life-sustaining substances synthesized, in addition to a range of plastics, lubricants, etc.

(*c*) *Propellant materials.* Some propellants for chemical rockets could be synthesized as in (*b*), but in some circumstances it may be necessary to set up mining operations purely for the purpose of obtaining propellants.

The above being some of our requirements it is interesting and important to consider what materials are likely to be found. On a planetary body with seas and atmosphere similar to Earth, it seems geologically sound to expect similar mineral deposits. However on an airless, waterless globe, or one nearly so, such as the Moon, the situation will be somewhat different. The lack of water will preclude the possibility of alluvial or sedimentary deposits and also organic deposits as we know them, due to the absence of terrestrial types of life. This condition at one stroke removes any hope of finding deposits comparable with those worked by some of the world's largest mines; unfortunately some of these are the very deposits which would be of the most value to the colony. It does in fact preclude the likelihood of finding any of the following:—

(*a*) Materials such as: coal, oil, limestone, chalk, ironstone, gannister, sandstone, brick clays, shale, slate, alluvial ores, bauxite, etc.

(*b*) Secondary enrichment of low grade primary ore-bodies.

(*c*) Ore deposits requiring sedimentary conditions for their emplacement, e.g., the banket gold ores of South Africa and the copper ores of Northern Rhodesia. (It is interesting to note that one suggested origin for these latter is in an epoch of ammoniacal seas on Earth.)

(*d*) Salt deposits such as those of Stassfurt, Cheshire, Yorkshire, etc., together with their associated gypsum and anhydrite.

It will be noticed that the above list includes the raw materials for making cement and mortar. This is not going to ease the civil engineering problem. Even so this list is far from complete; many other common materials will undoubtedly not be available (e.g., sand, gravel, etc.).

In addition to the above, there are other materials whose occurrence is doubtful, e.g., some of the metal ores, notably those of lead and zinc, often occur as intrusions into limestone country, so that the mode of occurrence, if any, of this particular type of ore-body in the absence of limestone remains to be seen.

Many useful minerals and rocks are, however, likely to be found. They include:—

Granite, syenite, basalt, and all other types of intrusive and extrusive rocks; gneisses, schists, etc., where these are derived from the metamorphism of

intrusive or extrusive rocks; kaolin (china clay), etc., as a result of the decomposition of granite, etc., by deep seated activity; all minerals of the types found in, for example, the granites and serpentines of Cornwall and similar places, i.e., the ores of arsenic, chromium, copper, gold, iron, lead, silver, tin, titanium, tungsten, zinc and uranium, together with their associated minerals, e.g., calcite, fluorspar, quartz, etc.

Of these the iron ores haematite and magnetite, the ore of chromium chromite, and the titanium ore ilmenite, are likely to be found massive, i.e., in such a condition that large pieces suitable for immediate reduction to the metal can be obtained directly without the need for separating large quantities of other minerals.

The other metals will undoubtedly be present in their ores in very much smaller quantities, usually well below 20 per cent., and some in almost trace amounts (gold has been worked in ore containing as little as five parts per million). The remainder of the ore will be wasted as far as the primary metal content is concerned, but in extra-terrestrial circumstances these "gangue minerals" may be very valuable, and in fact are often of great use even on Earth itself. Frequently minor constituents of the ore will be other valuable minerals.

Of the above ores, those of iron, chromium, tin, titanium, tungsten and other rarer metals not already mentioned will most probably occur as oxides. The ores of the other metals will most probably be sulphides. The main materials likely to occur with the metallic minerals are calcite $CaCO_3$, quartz SiO_2, fluorspar CaF_2, barytes $BaSO_4$. Even on Earth these materials are intrinsically valuable as fluxes in steel smelting, fillers for paper, etc.

In view of the lack of materials suitable for the manufacture of cement or mortar, it is likely that metals will play an even greater part in construction work than they do on Earth. It should be noted that "plastics" cannot help, since the raw materials for their manufacture, namely coal, oil, wood, milk, etc., will not be available.

One other point likely to cause difficulty is the shortage or absence of water. This, if not found free, may be very short indeed, since the mineral deposits likely to be found are not those which normally carry minerals with water in their make-up.

The minerals most likely to be found are shown in Appendices I and II. In Appendix I they are arranged in accordance with the acid radicles with which they are associated in nature; native elements are included for completeness. Appendix II gives an alphabetical list of some of the elements and their minerals which are most likely to be encountered, assuming that the rock formations, etc., to be found extra-terrestrially will in fact be similar to the igneous rocks, their metamorphic derivatives and associated intrusives as found on Earth.

Prospecting

On Earth the search for useful minerals has been going on for scores of centuries, and in the course of time what might be called a traditional method of prospecting has been evolved. Briefly stated this depends upon the fact that

most valuable minerals are heavier than the rocks, etc., in which they occur, and that they are usually more resistant to the disintegrating effects of wind and water.

Hence when a stream or river crosses a mineral outcrop, particles of mineral are carried along by the water until suitable conditions occur for heavy particles to collect in a particular spot; here a concentration of heavy minerals is found. Accordingly, if a river is followed through its course and suitable checks are made, those minerals occurring in its bed can be found. By noting the place in the stream or river where any particular mineral first appears as the river is followed downstream, then it is known that the source of the mineral must be immediately upstream from this point, and usually by a search of the banks can be found.

Many deposits have been located in this way, but as the more obvious and rich deposits have already been found, it has become necessary to develop other prospecting methods designed to locate orebodies which cannot be found in the traditional manner. These include:—

Core drilling of geologically likely situations, in which a hole is drilled into the rock to any required distance (the present limit is about 22,000 ft.), and a core brought up which is actually cut from the rocks *in situ*. In this way a deposit not outcropping at the surface can be located.

Geochemical prospecting takes advantage of the fact that during the weathering of rocks leaching takes place, so that redeposition in small quantities over a much wider area can occur. In addition to the weathering effects, areas of economic mineralization are sometimes surrounded by much larger areas where the mineralization has penetrated to a minor degree. If soil, subsoil or rock samples are taken and analysed for trace elements; local concentrations of any particular element which is unusual may indicate an orebody in the vicinity at no great depth.

Geobotanical prospecting is somewhat similar to the geochemical method. Vegetation samples are taken and analysed for trace elements, the results being interpreted in a similar fashion to those of geochemistry. Some types of vegetation grow more profusely where the concentration of a particular element is high, other species are completely inhibited. In the right circumstances a study of the vegetation can lead to the discovery of mineralization and this is often aided by aerial photography which readily shows changes in vegetation.

Geophysical prospecting covers all the physical methods used in locating mineral deposits. Probably the simplest of these is the *resistivity method* often used for finding the water table level by variations in the electrical conductivity of the strata. In dry conditions this can be used to locate orebodies. Some orebodies can act rather like an electric primary battery and the resulting E.M.F. can be detected; systems based on this phenomenon are known as *self potential methods*. *Gravity methods* show up strong concentrations of heavy or light minerals. *Seismic methods* give an indication of the nature of the rocks by the way in which shock waves, etc., are transmitted. *Magnetometric methods* detect areas of abnormally high or low

magnetic intensity; the former are usually associated with magnetite or ilmenite. *Electro-magnetic* methods operate by the effect of differing rock types upon search coils with alternating current flowing in them. Frequencies between 2½ kilocycles per second and 400 cycles per second are common. In many ways this method is similar in principle to that used in the mine detectors supplied to the armed forces, which are in fact devices for locating conductors. Recently a number of *radio-activity methods* for detecting concentrations of radio-active elements have been developed.

Of all these methods of prospecting only some can be used extra-terrestrially. Where water is absent there will be no chance of using the traditional method. In fact only on Mars does there seem any likelihood of this applying to any planet in this solar system other than Earth, with the exception of the giant planets where the force of gravity will in any case be too high to permit the landing of human beings, even supposing that propellants powerful enough to escape from such powerful gravitational fields were available. Geobotanical methods are also not likely to be applicable in most cases since they depend on plant life. Lack of weathering will greatly reduce the scope of geochemical methods. Magnetic, electro-magnetic, and radio-activity methods are often applied on Earth using fixed or moving wing aircraft flying at slow speeds and at low altitudes of 100 ft. to 2,000 ft. depending upon the method in use. In the absence of an atmosphere, the fuel which would have to be expended to achieve low altitude and slow speed flying would probably be prohibitive. The same methods can be used in conjunction with ground surveys, but in this case whether the survey is carried out on foot or using mechanical transport it will be a very much slower operation than would be an equivalent airborne project. In the case of magnetic methods, in the absence of a magnetic field even a large deposit of magnetite, etc., would not cause appreciable deflection of a compass needle unless very close indeed to the orebody in question. Hence it would be necessary to induce a field into the deposit electrically, and in this case the method would be classed with the electro-magnetic ones; under these conditions these latter will assume even greater importance than they do on Earth. The self potential method in most cases depends upon partial oxidation of some of the components of an orebody for its operation, and although it may find some applications, its use will be much more limited than on Earth.

From the preceding paragraph it will be seen that the methods available for extra-terrestrial prospecting are limited. Unfortunately they are in fact those which, whilst very effective in conjunction with data obtained by the traditional method, are not so easy to apply by themselves. However the lack of vegetation and weathering should make the identification of rock types easier than on Earth, and although lunar dust and similar deposits will be a hindrance, they will probably not cause so much difficulty as do soil, vegetation and weathering on Earth.

An important factor often overlooked when methods of prospecting are being considered, is that almost anywhere on Earth is inhabited, and although the inhabitants may not know the nature or importance of any particular mineral, ore or rock type, they will undoubtedly have noticed any physical

peculiarities of weight or colour which it may possess, and in fact prospectors get great help from local information. This information will of course be lacking elsewhere.

When considering the above points it must be remembered that the minerals being sought may in fact be the very ones which would be rejected as waste on Earth; e.g., calcite may be sought for the carbon dioxide which it contains and which can be easily liberated by heat.

It seems then that the pattern of extra-terrestrial prospecting will probably be somewhat as follows:

(1) Reconnaissance to obtain rock samples and pinpoint the positions at which their outcrops were discovered upon a map (the map may be the result of a rough survey carried out whilst collecting the samples, of aerial photographs taken whilst still in orbit, or better still a combination of the two).

(2) At base to identify the rock samples and draw out a suggested plan of the geology of the area.

(3) Based upon the plan just drawn make geophysical or other observations to amplify the existing knowledge of the structure.

(4) If the results from these last observations are similar to those which would be predicted from the geological plan in (2) proceed to the next stage. If not the geological plan must be modified in the light of the results obtained, and then further observations made on this basis to check the correctness of the new plan.

(5) On the basis of the above results and conclusions make detailed local searches for minerals.

(6) Test any orebearing outcrops found, by drilling (though lack of air and water will make it necessary to develop new techniques or considerably modify existing ones), driving of adits, sinking of prospect pits or trenches, or where justified by sinking prospect shafts and putting out crosscuts and drives from these.

(7) Reject or accept the prospect from the evidence found by the operations in (6). If it is accepted it then passes to the actual mining stage.

(8) If the detailed local search for minerals (5) fails but it is felt that the area in question still has possibilities, then more localized and detailed geophysical, geochemical or similar work must be carried out and the results of this confirmed or rejected by core drilling or similar methods. If an orebody is found then detailed investigations and decisions must be made as in (6) and (7). If nothing is found then the area should be abandoned and the search carried on elsewhere.

FOR SALE.—*Popular Astronomy*, January, 1947, to December, 1951 (50 issues). *Discovery*, November, 1944, to October, 1956 (missing: August, October, December, 1953, issues).

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THE DESCENT OF AN EARTH-SATELLITE THROUGH THE UPPER ATMOSPHERE

By D. G. King-Hele, M.A.

SUMMARY

A theoretical study is made of the flight path of an uncontrolled but aerodynamically stable satellite as it spirals down through the Earth's atmosphere from an initially circular orbit under the action of air drag. The Earth and its atmosphere are taken as spherically symmetrical. For altitudes above about 125 nautical miles a simple solution is found: the velocity of the satellite is independent of its size, shape, weight and initial altitude and equal to the orbital velocity appropriate to its current altitude; while its angle of descent, in radians, is twice its drag/weight ratio. Estimates of lifetime are also made.

Introduction

Now that Earth-satellites are emerging from the domain of fancy into the colder climate of reality, the citizens who live beneath their orbits will be asking sharper questions about the satellites' probable behaviour when, under the prolonged action of air drag, they come down to Earth. A 20-lb. satellite seems trivial enough, but the average householder might be somewhat daunted at the prospect of a 20-ton satellite, red-hot, landing in his garden. The likelihood of such a landing depends on the speed and angle of descent of the satellite at the altitudes where severe aerodynamic heating occurs; and this in turn depends on the past history of the satellite, on how it behaves as it spirals down through the upper atmosphere. In this paper it is shown that at altitudes above about 125 nautical miles simple and general formulæ for the velocity and angle of descent can be derived, if certain simplifying assumptions, stated below, are accepted.

Assumptions

The two main assumptions are:

(i) The Earth is spherical and non-rotating, with a spherically symmetrical atmosphere and gravitational field; and the satellite is initially in a circular orbit.

(ii) The satellite is uncontrolled and perfectly stable aerodynamically, remaining at zero incidence throughout its descent path. Thus its axis points in the direction of motion and the only force acting, other than gravity, is axial drag.

These assumptions are helpful simplifications in a first attack on the problem and it is difficult to guess how serious are the errors incurred. In practice the Earth's oblateness, or rather the atmosphere's oblateness, would impose a periodic variation on the axial drag. Also, unless the satellite was a sphere, its incidence would not be zero, but would vary periodically in a weathercock oscillation of low frequency; and superposed on this oscillation would be another resulting from the rotation of the atmosphere. Intuitively one would expect that these various disturbances would merely impose small perturbations on the simplified solution derived here, though there is always the risk that the perturbations may build up through resonance.

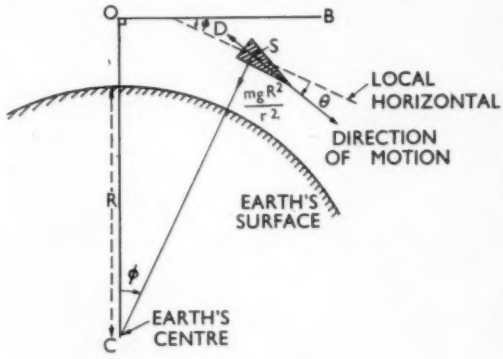

FIG. 1. Notation.

Analysis

Fig. 1 shows the notation. CO is a fixed line through the initial position of the satellite and S is its position at any later time t. The two forces acting on the satellite are drag D parallel to its axis, and weight mgR^2/r^2, where m is its mass, g the acceleration due to gravity at the Earth's surface, R the Earth's radius and r the distance CS between the satellite and Earth's centre. Let ϕ be the angle between CS and CO, let v be the satellite velocity, and let θ be its angle of descent, i.e. the inclination of its flight path to the local horizontal.

Then, resolving tangential and normal to the flight path, we have, as the dynamical equations of motion,

$$m\frac{dv}{dt} = \frac{mgR^2}{r^2}\sin\theta - D \quad .. \quad .. \quad .. \quad (1)$$

$$mv\frac{d}{dt}(\phi + \theta) = \frac{mgR^2}{r^2}\cos\theta, \quad .. \quad .. \quad .. \quad .. \quad (2)$$

since $(\phi + \theta)$ is the angle between the direction of motion and the fixed line OB. Equations (1) and (2) are supplemented by two kinematic equations expressing the radial and transverse velocities, namely

$$\frac{dr}{dt} = -v\sin\theta \quad .. \quad .. \quad .. \quad .. \quad (3)$$

$$r\frac{d\phi}{dt} = v\cos\theta. \quad .. \quad .. \quad .. \quad .. \quad (4)$$

If we assume that θ is small, so that $\sin\theta$ can be taken as θ and $\cos\theta$ as 1, and use equation (4) to eliminate t, equations (1)–(3) become

$$v\frac{dv}{d\phi} = \frac{gR^2\theta}{r} - \frac{rD}{m} \quad .. \quad .. \quad .. \quad .. \quad (5)$$

$$v^2\frac{d\theta}{d\phi} = \frac{gR^2}{r} - v^2 \quad .. \quad .. \quad .. \quad .. \quad (6)$$

$$\frac{dr}{d\phi} = -r\theta \quad .. \quad .. \quad .. \quad .. \quad (7)$$

Dividing equations (5) and (6) by equation (7) to eliminate ϕ:

$$v\frac{dv}{dr} = -\frac{gR^2}{r^2} + \frac{D}{m\theta} \quad .. \quad .. \quad .. \quad (8)$$

$$v^2\theta\frac{d\theta}{dr} = -\frac{gR^2}{r^2} + \frac{v^2}{r} \quad .. \quad .. \quad .. \quad .. \quad (9)$$

As long as θ is small, the velocity of the satellite is likely to be near the orbital speed appropriate to its altitude, $\sqrt{gR^2/r}$: for if v were very much less than orbital speed the satellite would promptly "fall out of its orbit," i.e. θ would no longer be small. So it is logical to make the substitution

$$v^2 = \frac{gR^2}{r} - \lambda \quad .. \quad .. \quad .. \quad .. \quad (10)$$

and to assume that

$$\lambda \ll \frac{gR^2}{r} \quad .. \quad .. \quad .. \quad .. \quad (11)$$

Eliminating v from equation (9) by means of equation (10)

$$\lambda = -r\theta\frac{d\theta}{dr}\left(\frac{gR^2}{r} - \lambda\right)$$

$$= -\frac{gR^2}{2}\frac{d}{dr}(\theta^2) \quad \text{to the first order} \quad .. \quad (12)$$

since $\lambda \ll gR^2/r$. Differentiating equation (10), we have

$$2v\frac{dv}{dr} = -\frac{gR^2}{r^2} - \frac{d\lambda}{dr} \quad .. \quad .. \quad .. \quad .. \quad (13)$$

Eliminating $v\,dv/dr$ from (8) and (13):

$$\frac{gR^2}{r^2} - \frac{2D}{m\theta} = \frac{d\lambda}{dr}$$

$$= -\frac{gR^2}{2}\frac{d^2}{dr^2}(\theta^2) \quad .. \quad .. \quad .. \quad .. \quad (14)$$

by equation (12). To solve equation (14) we assume first that since θ is small the right-hand side of (14) is almost negligible, so that

$$\theta \simeq \frac{2Dr^2}{mgR^2} = 2\frac{D}{mgR^2/r^2} = 2\frac{\text{drag}}{\text{weight}} \quad .. \quad .. \quad .. \quad (15)$$

Adopting the usual aerodynamic notation we now define the drag coefficient C_D by

$$C_D = \frac{D}{\frac{1}{2}\rho v^2 S} \quad .. \quad .. \quad .. \quad .. \quad (16)$$

where ρ is the density of the air around the satellite, and S is a reference area, taken as the maximum cross-sectional area of the satellite perpendicular to the direction of motion.

With this notation (15) becomes

$$\theta \simeq \rho r\left(\frac{SC_D}{m}\right)\left(\frac{v^2 r}{gR^2}\right) = \Delta\left(\frac{v^2 r}{gR^2}\right)\rho r \quad .. \quad .. \quad (17)$$

if we denote SC_D/m by Δ. Since C_D is constant at high Mach numbers in free-molecule flow, Δ may be taken as constant. Also v^2r/gR^2 is nearly constant by equations (10) and (11), while ρr varies enormously with altitude. So, using (17), we may write

$$\frac{d^2}{dr^2}(\theta^2) \simeq \Delta^2\left(\frac{v^2r}{gR^2}\right)^2\frac{d^2}{dr^2}(\rho r)^2 = \Delta^2\left(\frac{v^2r}{gR^2}\right)^2 f'' \quad .. \quad .. \quad (18)$$

where $f = (\rho r)^2$ and dashes denote differentiation with respect to r.

Equation (14) may be rewritten

$$\frac{2D}{m\theta} = \frac{gR^2}{r^2}\left\{1 + \frac{r^2}{2}\frac{d^2}{dr^2}(\theta^2)\right\}$$

and, on using (16) and (18) this becomes

$$\frac{\rho v^2}{\theta}\cdot\frac{SC_D}{m} = \frac{gR^2}{r^2}\left\{1 + \frac{r^2\Delta^2}{2}\left(\frac{v^2r}{gR^2}\right)^2 f''\right\}$$

or, since $(v^2r/gR^2) \simeq 1$,

$$\theta = \rho r\,\Delta\frac{v^2r}{gR^2}\left(1 - \frac{r^2f''\Delta^2}{2}\right) \quad .. \quad .. \quad .. \quad (19)$$

to the first order.

Also in equation (12) we may take

$$\frac{d}{dr}(\theta^2) = \Delta^2\left(\frac{v^2r}{gR^2}\right)^2 f',$$

so that, from (10) and (12)

$$v^2 = \frac{gR^2}{r}\left\{1 + \frac{rf'\Delta^2}{2}\right\} \quad .. \quad .. \quad .. \quad (20)$$

to the first order, on taking $v^2r/gR^2 = 1$ in the second (small) term.

Equations (19) and (20) are the first order solutions for θ and v.

In order to evaluate f' and f'' we must specify a "standard atmosphere," i.e. fix the variation of ρ with r. At present, however, the air density at high altitudes is not known at all accurately, and any chosen "standard atmosphere" will no doubt prove erroneous when the true values are known. The values chosen here for ρ between 100 and 200 n. miles altitude are from ref. 1 and are given in the Table overleaf. With these values of ρ, the quantity ρr can be approximated by the function

$$\rho r \simeq e^{2674\cdot160 - 236\cdot2804r \times 10^{-4} + 5\cdot195r^2 \times 10^{-13}} \quad .. \quad .. \quad (21)$$

the maximum error, for altitudes between 100 and 200 n. miles, being about 3 per cent., as the Table shows. Values of rf' and r^2f'', found by differentiating $f = (\rho r)^2$ as given by (21), are also included in the Table.

The Table provides values of rf' and r^2f'' for substitution in equations (19) and (20): it only remains to choose values for Δ. At altitudes above about 100 n. miles the mean free path of the air molecules is greater than the dimensions of the satellite and "free-molecule" aerodynamics must be used. In these conditions the drag coefficient C_D of the satellite, whatever its shape, may

TABLE I

THE VARIATION WITH ALTITUDE OF ρ, ρr, ETC.

Altitude n. miles	100	120	140	160	180	200
ρ lb./cu. ft.	471×10^{-12}	103×10^{-12}	$27{\cdot}9 \times 10^{-12}$	$8{\cdot}65 \times 10^{-12}$	$3{\cdot}18 \times 10^{-12}$	$1{\cdot}30 \times 10^{-12}$
ρr lb./sq. ft.	$10{\cdot}1 \times 10^{-3}$	$2{\cdot}23 \times 10^{-3}$	607×10^{-6}	189×10^{-6}	$69{\cdot}9 \times 10^{-6}$	$28{\cdot}8 \times 10^{-6}$
Approx. for ρr (21) ..	$10{\cdot}1 \times 10^{-3}$	$2{\cdot}29 \times 10^{-3}$	605×10^{-6}	188×10^{-6}	$67{\cdot}7 \times 10^{-6}$	$28{\cdot}7 \times 10^{-6}$
$-rf'$ ..	$55{\cdot}6 \times 10^{-3}$	$2{\cdot}61 \times 10^{-3}$	164×10^{-6}	$13{\cdot}9 \times 10^{-6}$	$1{\cdot}56 \times 10^{-6}$	$0{\cdot}235 \times 10^{-6}$
r^2f'' ..	31·6	1·35	$76{\cdot}7 \times 10^{-3}$	$5{\cdot}83 \times 10^{-3}$	578×10^{-6}	$75{\cdot}8 \times 10^{-6}$

conveniently be taken as 2 as long as the satellite's speed is much greater than the mean molecular speed. So $\Delta = SC_D/m = 2S/m$, where S is the satellite cross-sectional area in sq. feet and m is its mass in lb. Next we must set limits to S/m. If we exclude objects like metal-foil "balloons," and also exclude anything smaller than the U.S. 20-in. sphere of mass 20 lb., we may take 0·1 as an upper limit for S/m. As an example at the other end of the scale we may take a 10,000-lb. satellite with a cross-sectional area of 50 sq. ft. ($\simeq$ 8 ft. diameter), giving $S/m \simeq 0{\cdot}005$. Thus we assume here that Δ lies between 0·01 and 0·2.

If the simplest solutions for v and θ, namely

$$v^2 = \frac{gR^2}{r} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (22)$$

and

$$\theta = \rho r \Delta, \quad .. \quad .. \quad .. \quad .. \quad (23)$$

are to be reasonably accurate, the terms $r^2f''\Delta^2/2$ and $rf'\Delta^2/2$ in equations (19) and (20) must be small. The Table shows that $|rf'|$ is always less than 1/300 of r^2f'', and so the error in equation (22) will always be much less than the error in equation (23). If equation (23) is to hold with, say, 1 per cent. accuracy, $r^2f''\Delta^2$ must be less than 0·02, and using the values of r^2f'' in the Table, we find that the maximum permissible values for Δ are 0·025 at 100 n. miles altitude, 0·12 at 120 n. miles altitude and 0·51 at 140 n. miles. Thus the values of θ given by equation (23) are in error by less than 1 per cent. down to about 127 n. miles altitude if $\Delta = 0{\cdot}2$ or about 90 n. miles if $\Delta = 0{\cdot}01$. Under the same conditions the values of v given by (22) are in error by less than 0·005 per cent. Equation (23) can be re-written in the same form as (15).

$$\theta = 2\,\frac{\text{drag}}{\text{weight}},$$

for

$$\rho r \Delta = \frac{\rho r S C_D}{m} = 2\,\frac{\frac{1}{2}\rho v^2 S C_D}{mgR^2/r^2},$$

since

$$v^2 = gR^2/r.$$

Results

At altitudes where equations (22) and (23) hold, i.e. above about 90 n. miles altitude if Δ is small ($\simeq$ 0·01) or above about 125 n. miles if Δ is large ($\simeq$ 0·2),

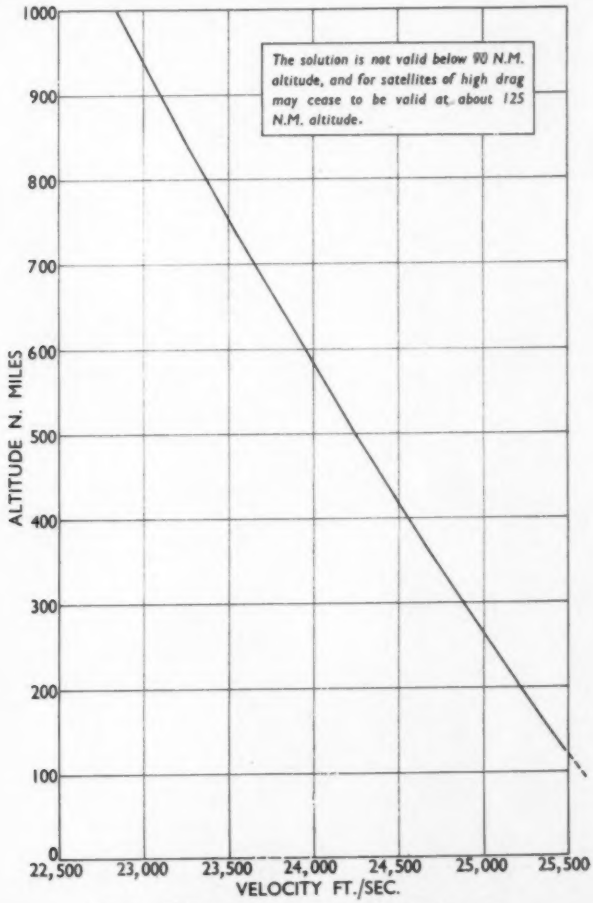

FIG. 2. Velocity of a satellite during its descent.

the speed of the satellite at any altitude is equal to the orbital speed appropriate to that altitude, $\sqrt{gR^2/r}$, and the angle of descent θ (the inclination of the flight path to the local horizontal) is very nearly equal to $\rho r\Delta$, or twice the drag/weight ratio, the maximum error being 1 per cent.

Fig. 2 shows how the speed varies with altitude, and Fig. 3 gives the variation of θ with altitude between 200 and 100 n. miles, accepting the values of ρ given in the Table. At 130 n. miles altitude the speed has reached 25,460 ft./sec., while the angle of descent lies between 0·00066° for $\Delta = 0{\cdot}01$ and 0·0132° for $\Delta = 0{\cdot}2$.

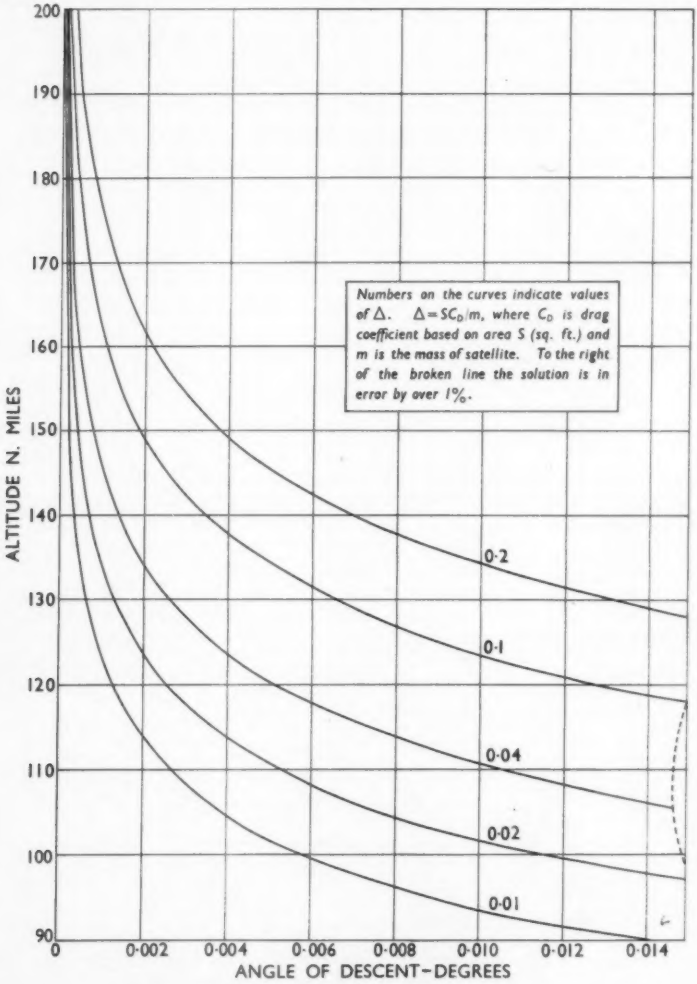

FIG. 3. Variation of angle of descent with altitude.

The contrast between Fig. 2 and Fig. 3 is interesting. The satellite's speed is independent of its shape, size and weight, i.e. is independent of Δ, whereas its angle of descent is directly proportional to Δ and hence inversely proportional to the mass/area ratio m/S. The increase in air density as the satellite descends causes a rapid increase in the angle of descent, which is proportional to the density. But the increase in drag as the satellite descends does not reduce the

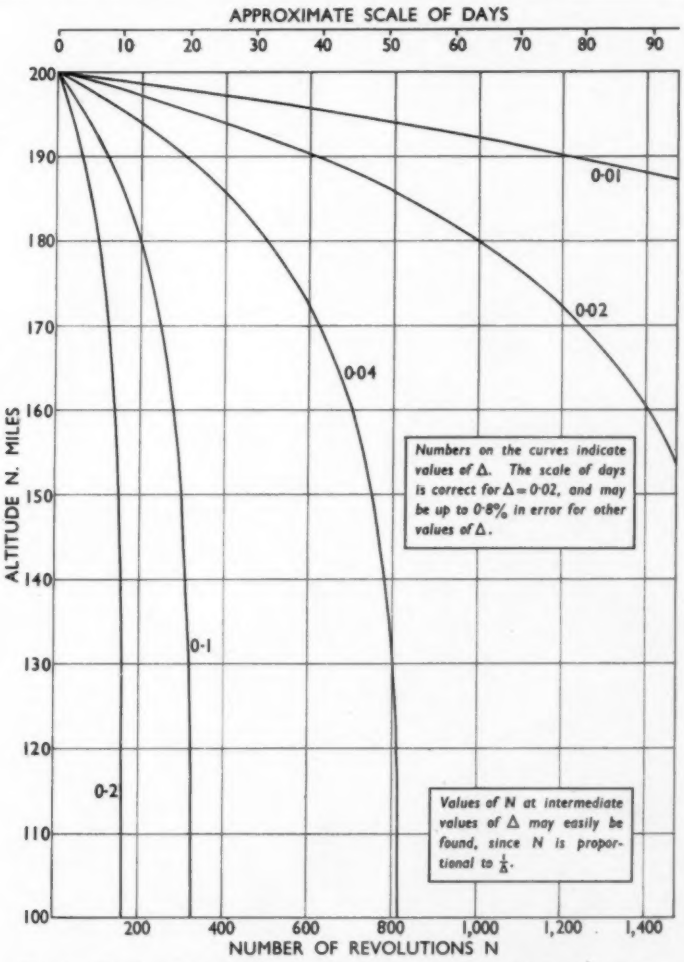

FIG. 4. Number of revolutions made in descending from 200 n.m. altitude.

speed, as might be expected. On the contrary, the speed goes on rising steadily, because the angle of descent always adjusts itself so that half the potential energy accruing from loss in altitude is absorbed by the drag while the other half goes to increase the kinetic energy. This increase in speed as the satellite descends, which, of course, implies a shorter period of revolution, does not go on indefinitely. The velocity reaches a maximum (at about 80 n. miles altitude when $\Delta = 0{\cdot}02$) and then falls rapidly.

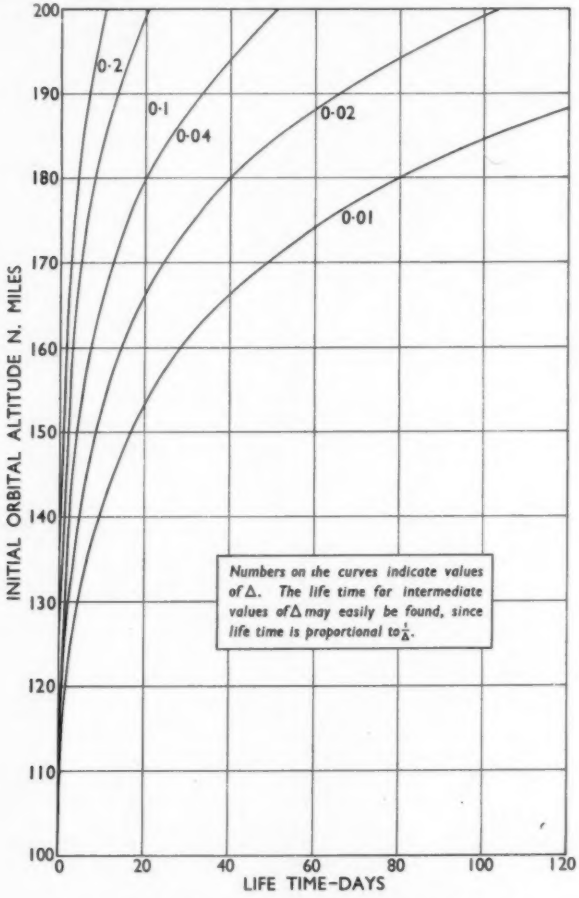

FIG. 5. Variation of lifetime with initial orbital altitude.

Once the angle of descent θ is known, the number of revolutions N made by the satellite in dropping from any given initial altitude can easily be found by integration of equation (7) in the form

$$dN = \frac{d\phi}{2\pi} = -\frac{dr}{2\pi r\theta}$$

which shows incidentally that N is proportional to $1/\theta$, i.e. to $1/\Delta$, for given r. The number of revolutions performed by satellites having various values of Δ and an initial altitude of 200 n. miles is shown in Fig. 4, a scale of days also

being included. From this graph the number of revolutions completed between any two altitudes (in the 100–200 n. miles bracket) can be found: e.g. for $\Delta = 0{\cdot}04$, $N \simeq 750$ revolutions at 150 n. miles, and $N \simeq 810$ at 120 n. miles, i.e. about 60 revolutions are made in descending from 150 to 120 n. miles altitude. It is clear from Fig. 4 that very few more revolutions will be performed below 100 n. miles; so the time taken to drop to 100 n. miles can conveniently be taken as the lifetime of the satellite. Lifetime, so defined, is plotted against initial altitude for various Δ in Fig. 5. The lifetimes plotted in Fig. 5 are in fair agreement with those given in reference 2, which were derived in a different way.

The values of v and θ given in Figs. 2 and 3 are those obtained from the simplest solutions, equations (22) and (23). v and θ can be found with similar accuracy down to an altitude some 7 n. miles lower by using equations (20) and (19) instead of (22) and (23). The lower limit of altitude can be pushed a mile or two further still if the second-order approximations for v and θ are used. These are found to be

$$v = \sqrt{\frac{gR^2}{r}} \left\{ 1 + \frac{rf'\Delta^2}{4} - \frac{r^3\Delta^4}{4}\left(f'f'' + ff'''\right)\right\}$$

$$\theta = \rho r \Delta \left\{ 1 - \frac{r^2 f''\Delta^2}{2} + \frac{r^4\Delta^4}{2}\left(\frac{3}{2}f'' + 2f'f''' + ff''''\right)\right\}$$

if certain terms which can be shown to be small are ignored.

REFERENCES

(1) Handbook of supersonic aerodynamics. U.S. Navord Report 1488, April, 1950.
(2) N. V. Petersen, "Lifetimes of satellites in near-circular and elliptic orbits," *Jet Propulsion*, **26**, 341, May, 1956.

LIST OF SYMBOLS

Symbol		*Unit*
C_D	= drag coefficient $= D/\frac{1}{2}\rho v^2 S$, taken as 2.	
D	= drag	pdls
e	= 2·71828 . . .	
f	$= (\rho r)^2$	lb.²/ft.⁴
g	= acceleration due to gravity at Earth's surface, taken as 32·2	ft./sec.²
m	= mass of satellite	lb.
N	= number of revolutions.	
r	= distance of satellite from Earth's centre	ft.
R	= Earth's radius (taken as $10{,}800/\pi$ n. miles, or $20{\cdot}9015 \times 10^6$ ft.).	
S	= cross-sectional area of satellite, in plane perpendicular to its axis	sq. ft.
t	= time	sec.
v	= velocity of satellite	ft./sec.
Δ	$= SC_D/m$	ft.²/lb.
θ	= angle of descent of satellite = inclination of flight path to local horizontal.	
λ	$= gR^2/r - v^2$	ft.²/sec.²
π	= 3·14159 . . .	
ρ	= air density	lb./ft.³
ϕ	= angular travel of satellite relative to Earth's centre (*see* Fig. 1).	

Acknowledgement

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ORIENTATION IN SPACE

By M. Vertregt

1. Introduction

Orientation in space must be based on different principles from orientation on Earth, for on Earth we have several means of orientation which do not exist in space. On Earth we have many landmarks, e.g. mountain-tops, towers and high buildings, which we can use; on the sea we have lighthouses to guide the mariner, and on the open sea, where no landmarks are visible, the compass will show the direction in which we are heading. In the air, when the atmosphere is cloudy and misty, the pilot of an aircraft can find his way by means of the modern inventions of radar and radio-sounding.

When nothing on the surface of the Earth is visible and radio stations are too far off, we can use the sun or the stars to determine our position.

Out in space, there are no nearby landmarks to give us an approximate idea of our position; there is no sense of movement, nor of direction. On a ship or a 'plane we feel the tremor of the motors, but travelling in space in free flight gives us the nearest approximation to absolute rest. There is no discernible motion outside the spaceship, even if one travels at a speed of tens of miles per second. One seems to hang motionless in a motionless void. The only movement we can observe is the slow rotation of the spaceship, which only helps to confuse us as to the real direction of our flight.

Only after days of travelling at an enormous speed can the practised eye observe a slight shift in the position of a few of the myriads of luminous points which seem to be fixed immovably on an absolutely motionless globe.

If we want to determine the speed and the direction of our flight with some degree of exactness, direct observation is of no use; we need observations with accurate instruments, but even with the most accurate instruments an error in position of several times the diameter of the Earth will be normal.

2. Systems of Co-ordinates

If we consider the stars placed on the surface of a globe, which has its centre in the centre of the Earth, we can extend the plane of the equator of the Earth until it intersects this globe. This line of intersection is called the celestial equator. The points of intersection of the axis of the Earth with this globe are the celestial poles.

We have to determine a point on the celestial equator as the zero-point. Astronomers choose for this zero-point the point which the centre of the Sun occupies at the Vernal Equinox.

The place of a star S (Fig. 1) is then determined by two co-ordinates: OA, called the Right Ascension α and AS, called the Declination δ.

This frame of reference is adapted to terrestrial circumstances and is not the most suitable for space-travelling. The system is based on the direction of the Earth's axis, and this direction has no importance once we have left the Earth. Furthermore, this direction is not constant. In 26,000 years the celestial pole describes a full circle on the star-globe with a radius of 23·5°. As

a consequence of this the Vernal Equinox is not a fixed point on the equator, but advances each year about 50·3″, or one degree in about seventy years. Also the angle of the axis with the plane of the Earth's orbit is not constant. As a result of this the places of the stars in this frame of reference change continually. The plane of reference most suited for space-travel, if for the present we consider only interplanetary voyages, is the plane of the Earth's orbit around the Sun. Except for very small perturbations, caused by the Moon and the planets, this plane has a fixed position among the stars, and therefore it is much better suited for space travel.

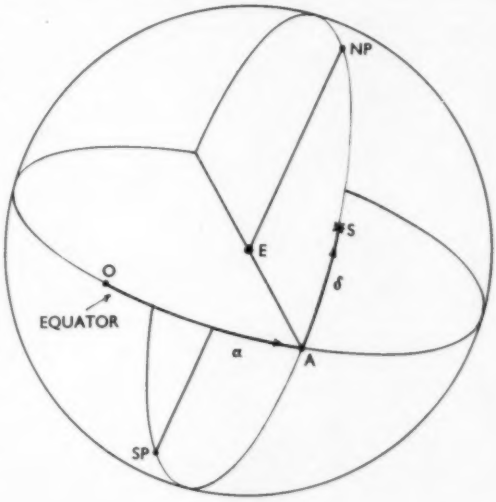


FIG. 1

The planes of the orbits of the other planets do not differ very much from the Earth's plane, except in the case of Pluto. The angles which the orbits make with the Earth's orbit are as follows:—

Mercury	.. ..	7°	Saturn	.. ..	2·5°
Venus	.. ..	3·5°	Uranus	.. ..	1°
Mars ..	.. ..	2°	Neptune	.. ..	2°
Jupiter	.. ..	1°	Pluto	.. ..	17°

Thus, the logical thing to do is to accept the plane of the Earth's orbit as the plane of reference, with the Sun in the centre (Fig. 2).

We call the arc OA, the *longitude* λ and the arc AS the *latitude* B of star S.

The great circle which is the intersection of the extended plane of the Earth's orbit with the celestial globe, is called the *ecliptic*. As in the system of Right Ascension and Declination the Vernal Equinox, which is the point of intersection of the equator and the ecliptic, is the zero-point.

The angle ϵ between the planes of the equator and of the ecliptic is 23° 27′, which is not constant and decreases about 0·47″ per year.

To correct the co-ordinates of one system into the co-ordinates of the other system, we have the formulae:

$$\sin \beta = -\cos \delta \,.\, \sin \alpha \,.\, \sin \epsilon + \sin \delta \,.\, \cos \epsilon \quad .. \quad .. \quad (1)$$

$$\cos \lambda = \frac{\cos \delta \,.\, \cos \alpha}{\cos \beta} \quad .. \quad .. \quad .. \quad .. \quad .. \quad (2)$$

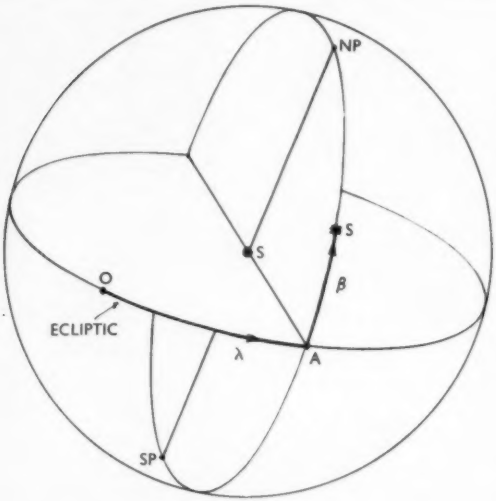


FIG. 2

Now we saw that the Vernal Equinox, which is used in both systems as the zero-point is not a fixed point.

As the Vernal Equinox is inherent to the position of the Earth, this point is of no importance outside the Earth, and therefore for space travel we can look for another point which does not possess the disadvantages of inconstancy.

A bright star situated near the ecliptic can serve much better, e.g. α Leonis (or Regulus), the brightest star in the constellation of Leo. The latitude of Regulus is $+ 0° 27'$, its longitude $168° 59'$, its magnitude 1·3, and its parallax is 0·049″. The proper motion of Regulus is 0·25″. Thus Regulus suffices the conditions for a fixed point, and it would be practical to define the zero-point in this system as the point of intersection between the great circle through the celestial pole and Regulus, and the ecliptic.

3. Determination of the Position of the Spaceship

The position of a spaceship is determined by its heliocentric longitude and latitude according to the frame of reference, described in the foregoing paragraph, and its distance from the Sun.

To obtain the first two data it is only necessary to observe the apparent longitude and latitude of the Sun among the fixed stars, as seen from the spaceship. For the third we need a planet, whose heliocentric longitude and latitude and whose distance from the Sun (for any given moment) must be known and whose apparent longitude as seen from the spaceship is found by observation. The space-traveller will need an Astronautical Almanac, where the positions of the planets and their distances from the Sun can be found in astronomical tables.

In Fig. 3, S is the Sun, P the planet and R the spaceship. The broken lines are situated in the plane of the ecliptic. P_1 and R_1 are the vertical projections of P and R on this plane.

From the Astronautical Almanac we get the following data:

Longitude of planet	λ_2.
Latitude of planet	β_2.
Distance of planet from the Sun ..	r_2.

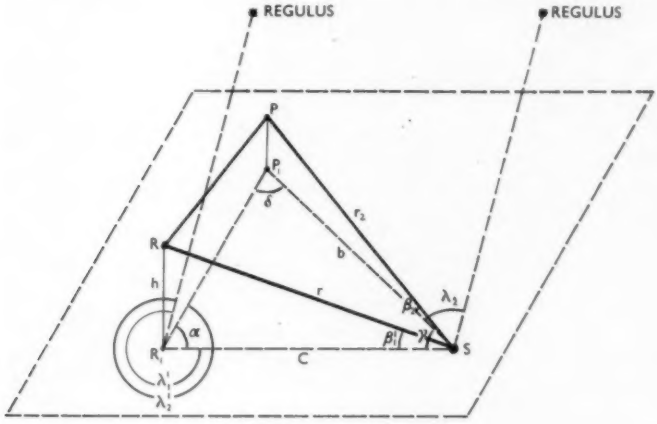


FIG. 3

From observation we get the following data:

Apparent longitude of the Sun	.. ..	λ_1'.
Apparent latitude of the Sun	.. ..	$-\beta_1'$.
Apparent longitude of planet	.. ..	λ_2'.

From Fig. 3 we see that:

$$b = r_2 \cos \beta_2. \qquad c = b \cdot \frac{\sin \delta}{\sin \alpha}$$

$$\delta = 360^\circ + \lambda_2 - \lambda_2'. \qquad \alpha = \lambda_2' - \lambda_1'.$$

$$r = \frac{c}{-\cos \beta_1'} = \frac{c}{\cos \beta_1'}.$$

Thus:

$$r = r_2 \cdot \frac{\cos \beta_2}{\cos \beta_1'} \cdot \frac{\sin (\lambda_2 - \lambda_2')}{\sin (\lambda_2' - \lambda_1')} \quad .. \quad .. \quad .. \quad (3)$$

If h is the distance of the spaceship from the plane of the ecliptic, then:

$$h = -r \sin \beta_1' \quad .. \quad .. \quad .. \quad .. \quad (4)$$

β_1' and h are counted positive for angles and distances north, and negative for angles and distances south of the plane of the ecliptic.

By taking the Earth as the planet of reference these equations can be simplified, because for the Earth the heliocentric latitude β_2 is always zero. If the spaceship travels in the plane of the ecliptic the apparent latitude of the Sun will also be zero.

Then, equation (3) becomes:

$$r = r_2 \cdot \frac{\sin (\lambda_2 - \lambda_2')}{\sin (\lambda_2' - \lambda_1')} \quad .. \quad .. \quad .. \quad (5)$$

The observation of the apparent longitudes and latitudes of the Sun and the planet is done by measuring the angles of their apparent positions with some nearby stars, the co-ordinates of which can be found in a table.

The measuring of these angles from a spaceship will be no easy matter. The navigator on a ship at sea has an uninterrupted view of the whole starry sky from the bridge and, except in rough weather, the movements of the ship will be small. In favourable circumstances angles can be measured with a sextant to an accuracy of about 0·5′. In science-fiction one often sees pictures of spaceships with a roomy cupola of some transparent plastic, giving a beautiful view of almost the whole celestial globe. But as every interruption of the hull-structure causes a weak place, I think that the first spaceships will only need a few small portholes. Furthermore one must not imagine that a spaceship will be as motionless as a ship on a smooth sea. On Earth every motion tends to die out owing to friction with the land, water, or air. In space there is no friction, so every motion will continue indefinitely until it is opposed by another motion.

Therefore, besides the forward motion, the spaceship will always show a rotation about some axis. Every shifting of a mass inside the ship, say a passenger moving from one side to the other, will cause a rotation, and this rotation will be the quicker the smaller the mass of the spaceship.

Even the slowest rotation will cause great difficulties for visual measurement of the angles through the small port-holes, and it will not be easy to suppress this rotation altogether. If this is done by auxiliary jets, we need three jets for the three perpendicular axes of the ship, and the jets must be able to give very small thrusts of the order of magnitude of some ounces, to kill the last remnants of rotation. It will require some tricky manœuvring to bring a rotation temporarily to a complete standstill.

Another method of stopping rotation is to give a counter-rotation by means of a gyroscope, but a gyroscope has the drawback that when the power is cut off and the disc runs out owing to friction in the bearings, the original rotation of the ship will be restored.

A. C. Clarke has jokingly proposed to put the spinning gyroscope outside as the only remedy against this evil, but luckily in practice this drastic measure will not be necessary. If the gyroscope runs in a vacuum and the friction in the bearings is small, it will take an appreciable time before the rotation is restored.

The best method would be to make photographs of the Sun and of the planet among the stars. These photographs would have to show only a small part of the heavens, say 10 degrees square.

In a strip extending 20 degrees north and south of the ecliptic, there are seven stars of magnitude one or brighter, i.e.:

α Scorpio (Antares)
α Virginis (Spica)
α Leonis (Regulus)
β Geminorum (Pollux)
α Canis Minoris (Procyon)
α Orionis (Betelgeux)
α Tauri (Aldebaran)

Furthermore there are eleven stars of the second magnitude, 56 stars of the third magnitude and about 650 stars of the fourth and fifth magnitudes. Thus a photograph 10 degrees square will, on average, show about seven stars of the fourth magnitude or brighter, which is sufficient for an accurate determination of the position of the Sun and of the planet.

The time of exposure of the photographs will, of course, be much shorter than for an exposure from the surface of the Earth, for we will have the full benefit of the total radiation of the star, including the very active ultra-violet part of the spectrum, which is absorbed by the terrestrial atmosphere. If necessary, electronic image intensifiers can be used to make instantaneous photographs of stars of the fifth magnitude.

The photographic magnitude of the stars will deviate much from the visual magnitudes, owing to the ultra-violet radiation. But as the "seeing" in space is always 100 per cent. the diameters of the images on the photographic plate will always be equal if standard equipment and equal times of exposure are used. The star-maps used in the spaceship will, of course, show the photographic magnitudes.

Taking photographs of the Sun and adjacent stars will require a special technique, and a filter of varying opaqueness which screens off most of the light of the Sun but which lets through the undiminished light of the stars will have to be used. Instead of the Sun a planet can be used, though the calculations in this case become a little more complicated.

3 (1). *Example.*

Assume that on October 10, at 10 h. G.M.T., in 2056 A.D. on spaceship X two photographs are taken, one of the Sun (Fig. 4) and one of the Earth (Fig. 5). Measurement of the plates gives the following:

Apparent Regulus-longitude of the Sun .. $\lambda_1' = 92° 6'$.
Apparent latitude of the Sun .. .. $\beta_1' = +8° 54'$.
Apparent Regulus-longitude of the Earth .. $\lambda_2' = 137° 42'$.

In the *Astronautical Almanac* for the year 2056 A.D., October 10, 10 h. G.M.T., we find the following heliocentric co-ordinates of the Earth:

Regulus-longitude .. .. $\lambda_2 = 227° 0'$
latitude .. .. $\beta_2 = 0°$
Distance from the Sun .. $r_2 = 1{\cdot}496 \cdot 10^8$ km.

Applying equation (3) we find:

$$r = 1{\cdot}496 \cdot \frac{\cos 0°}{\cos 8° 54'} \cdot \frac{\sin (227° 0' - 137° 42')}{\sin (137° 42' - 92° 6')} = 2{\cdot}118 \times 10^8 \text{ km.}$$

$$h = -2{\cdot}118 \times 10^8 \cdot \sin 8° 54' = -3{\cdot}278 \times 10^7 \text{ km.}$$

The heliocentric co-ordinates of the spaceship are:

Regulus-longitude $\lambda = \lambda_1' + 180° = 272° 6'$
latitude $\beta = -\beta_1' = -8° 54'$.

In the *Almanac* we find furthermore that the Regulus-longitudes of the

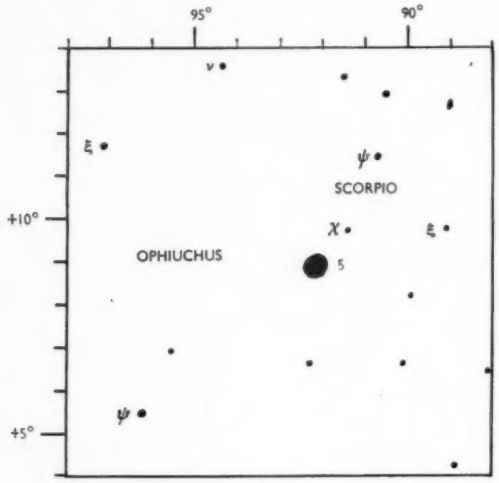


FIG. 4

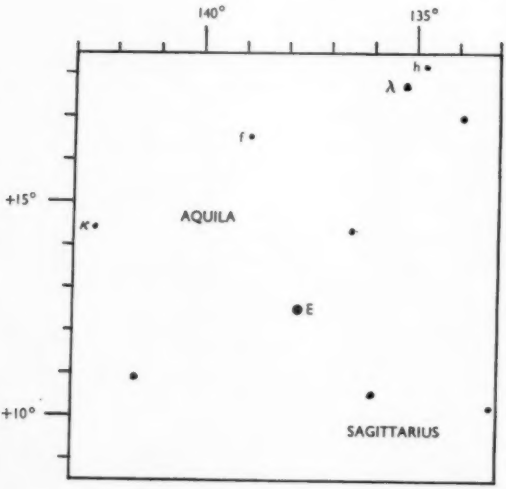


FIG. 5

nearer planets at this time are:

Mercury ..	28° 42′
Venus ..	135° 48′
Mars ..	283° 12′.

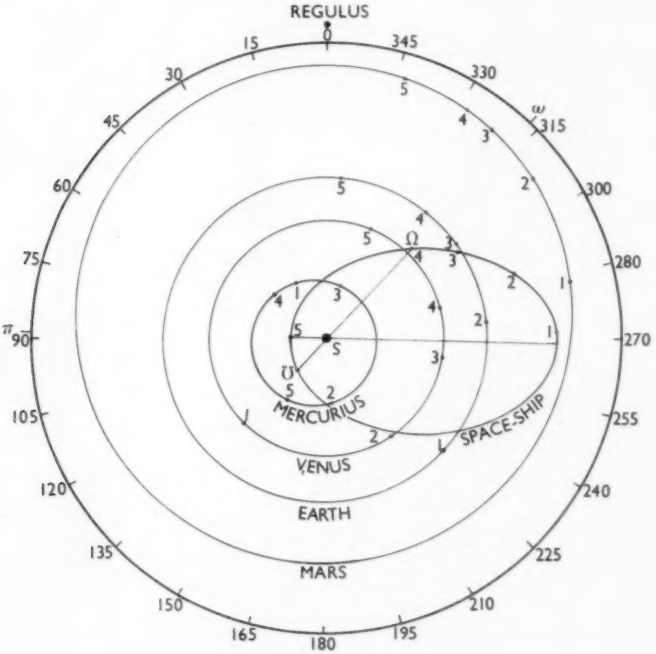
From this we can make a map of the position of the spaceship relative to the planets, see Fig. 6 (position (1)).

4. Determination of the Shape of the Orbit

If we know the positions of the spaceship on three consecutive dates, we can compute the orbit of the ship, assuming that it moves in free flight. This is the celebrated problem of Gauss, but whereas Gauss had only three positions of the minor planet Ceres from which to compute the entire orbit, including its perturbations from other planets, we can take our bearings every day and compare our real positions with the calculated positions and thus correct our orbit day by day. If we assume that our ship is in mid-space and that no planets are nearby, we can neglect the attraction of the planets and consider our ship to describe a pure Keplerian orbit.

If we assume that the spaceship is following an ellipse around the Sun, then:

$$r_1 = \frac{a\,(1-\epsilon^2)}{1+\epsilon\cos\theta} \quad .. \quad .. \quad .. \quad .. \quad (6)$$



POSITION	DATE
1	OCT. 10 2056
2	NOV. 29 2056
3	DEC. 28 2056
4	JAN. 12 2057
5	FEB. 13 2057

FIG. 6

where:

r_1 is the radius vector of the ship at position 1.

a is the semi-major axis of the ellipse.

ϵ is the eccentricity of the ellipse.

θ is the angular perihelion distance at position 1.

For the second observation we find likewise:

$$r_2 = \frac{a(1-\epsilon^2)}{1+\epsilon\cos(\theta+\phi_2)} \quad .. \quad .. \quad .. \quad (7)$$

where:

r_2 is the radius vector of the ship at position 2.

ϕ_2 is the angular distance between position 1 and position 2.

For the third observation we find:

$$r_3 = \frac{a(1-\epsilon^2)}{1+\epsilon\cos(\theta+\phi_3)} \quad .. \quad .. \quad .. \quad (8)$$

where:

ϕ_3 is the angular distance between position 1 and position 3.

From (6) and (7) we get:

$$r_1 + r_1 . \epsilon\cos\theta = r_2 + r_2 . \epsilon\cos(\theta+\phi_2) \quad .. \quad .. \quad (9)$$

From (6) and (8) we get:

$$r_1 + r_1 . \epsilon\cos\theta = r_3 + r_3 . \epsilon\cos(\theta+\phi_3) \quad .. \quad .. \quad (10)$$

By eliminating ϵ from (9) and (10) we get:

$$\frac{r_2 - r_1}{r_1\cos\theta - r_2\cos(\theta+\phi_2)} = \frac{r_3 - r_1}{r_1\cos\theta - r_3\cos(\theta+\phi_3)} \quad .. \quad (11)$$

or:

$$r_1(r_2-r_3)\cos\theta - r_2(r_1-r_3)\cos(\theta+\phi_2) + r_3(r_1-r_2)\cos(\theta+\phi_3) = 0 \quad (12)$$

We call:

$$r_1(r_2-r_3) = \mathrm{A}.$$
$$r_2(r_1-r_3) = B.$$
$$r_3(r_1-r_2) = C.$$

Then (12) becomes:

$$A\cos\theta - B\cos(\theta+\phi_2) + C\cos(\theta+\phi_3) = 0 \quad .. \quad .. \quad (13)$$

or:

$$A\cos\theta - B(\cos\theta . \cos\phi_2 - \sin\theta . \sin\phi_2) + C(\cos\theta . \cos\phi_3 - \sin\theta . \sin\phi_3) = 0 \quad .. \quad .. \quad (14)$$

From which:

$$\tan\delta = \frac{A - B\cos\phi_2 + C\cos\phi_3}{-B\sin\phi_2 + C\sin\phi_3} \quad .. \quad .. \quad .. \quad (15)$$

When ψ_1, ψ_2 and ψ_3 are the angular perihelion distances of positions 1, 2 and 3, then:

$$\psi_1 = \theta$$
$$\psi_2 = \theta + \phi_2$$
$$\psi_3 = \theta + \phi_3.$$

From (10) we get:

$$\epsilon = \frac{r_3 - r_1}{r_1\cos\psi_1 - r_3\cos\psi_3} \quad .. \quad .. \quad .. \quad (16)$$

And from (8):

$$a = \frac{r_3(1+\epsilon\cos\psi_3)}{1-\epsilon^2} \quad .. \quad .. \quad .. \quad .. \quad (17)$$

4 (1) *Example.*

From the following figures of the three positions of our spaceship we can compute the orbit.

Positions:	1	2	3
Dates ..	Oct. 10, 10 h. 2056	Nov. 29, 2 h. 2056	Dec. 28, 7 h. 2056
Regulus-longitude:	$\lambda(1) = 272° 6'$	$\lambda(2) = 289° 18'$	$\lambda(3) = 303° 30'$
Latitude:	$\beta(1) = -8° 54'$	$\beta(2) = -5° 54'$	$\beta(3) = -2° 48'$
Distance from the Sun:	$r_1 = 2{\cdot}118 \times 10^8$	$r_2 = 1{\cdot}823 \times 10^8$	$r_3 = 1{\cdot}436 \times 10^8$ km.

Firstly we compute the angular distances of the three positions. We saw that ϕ_2 is the angular distance between positions 1 and 2, and ϕ_3 is the angular distance between positions 1 and 3. Then, according to a well-known equation from spherical trigonometry:

$$\cos \phi_2 = \sin \beta(1) \,.\, \sin \beta(2) + \cos \beta(1) \cos \beta(2) \cdot \cos (\lambda(1) - \lambda(2)) \quad .. \quad (18)$$

or:

$$\cos \phi_2 = \sin 8° 54' \,.\, \sin 5° 54' + \cos 8° 54' \cdot \cos 5° 54' \cos (272° 6' - 289° 18').$$

From which:

$$\phi_2 = 17° 18'.$$

In the same manner we find:

$$\phi_3 = 31° 48'.$$

Now:

$$A = 2{\cdot}118\,(1{\cdot}823 - 1{\cdot}436) \times 10^8 \text{ km.} = 0{\cdot}8196 \times 10^8 \text{ km.}$$
$$B = 1{\cdot}823\,(2{\cdot}118 - 1{\cdot}436) \times 10^8 \text{ km.} = 1{\cdot}244 \times 10^8 \text{ km.}$$
$$C = 1{\cdot}436\,(2{\cdot}118 - 1{\cdot}823) \times 10^8 \text{ km.} = 0{\cdot}4236 \times 10^8 \text{ km.}$$

And according to (15):

$$\tan = \theta \frac{0{\cdot}8196 - 1{\cdot}244 \cdot \cos 17° 18' + 0{\cdot}4236 \,.\, \cos 31° 48'}{-1{\cdot}244 \,.\, \sin 17° 18' + 0{\cdot}4236 \,.\, \sin 31° 48'} = 0{\cdot}0567.$$

or:

$$\theta = \psi_1 = 183° 15'.$$

and:

$$\psi_2 = \theta + \phi_2 = 183° 15' + 17° 18' = 200° 33'.$$
$$\psi_3 = \theta + \phi_3 = 183° 15' + 31° 48' = 215° 3'$$

From (16):

$$\epsilon = \frac{1{\cdot}436 - 2{\cdot}118}{2{\cdot}118 \cos 183° 15' - 1{\cdot}436 \cos 215° 3'} = 0{\cdot}7280.$$

From (17):

$$a = \frac{2{\cdot}118 \times 10^8\,(1 - 0{\cdot}7280 \cos 183° 15')}{1 - 0{\cdot}7280^2} \text{ km.} = 1{\cdot}232 \times 10^8 \text{ km.}$$

5. Determination of the Position of the Orbit

After having thus calculated the major axis and the eccentricity of the ellipse and the positions of our spaceship on this ellipse, we have to determine the position of this ellipse relative to our frame of reference.

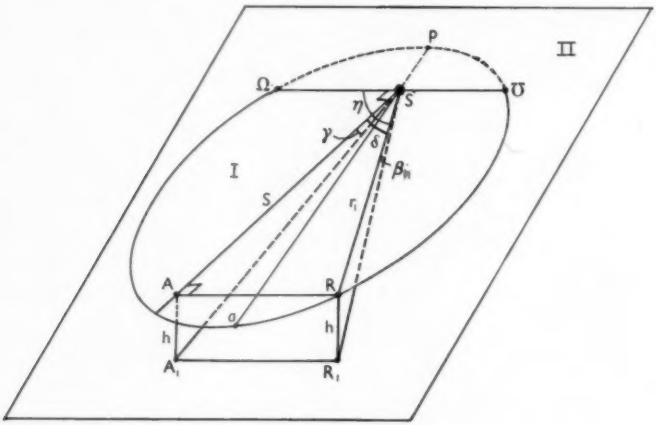


FIG. 7

Therefore we must know the angle of the plane of the ellipse with the plane of the ecliptic, the position of the perihelion and the position of the nodes (which are the points of intersection of the ellipse with the plane of the ecliptic).

In Fig. 7 the plane of the ellipse is designated by I and the plane of the ecliptic by II. Ω and $\mathfrak{V}$ are the ascending and the descending nodes. S is the Sun, R is position 1 of the spaceship.

$$SA \pm \Omega - \mathfrak{V}$$
$$AR \pm SA$$
$$SR = r_1$$
$$SA = s.$$

A_1 and R_1 are the vertical projections of A and R on plane II.

η is the angle between the radius-vector of position 1 and the ascending node.

$-\beta(1)$ is the apparent latitude of the Sun.

p is the perihelion.

a is the aphelion.

Then:

$$h = r_1 \sin \beta(1)$$
$$s = r_1 \cos \delta.$$
$$\delta = \eta - 90°$$

or:

$$s = r_1 \sin \eta.$$

Then:

$$\sin \gamma = \frac{h}{s} = \frac{r_1 \sin \beta(1)}{r_1 . \sin \eta} = \frac{\sin \beta(1)}{\sin \eta} \quad .. \quad .. \quad .. \quad (19)$$

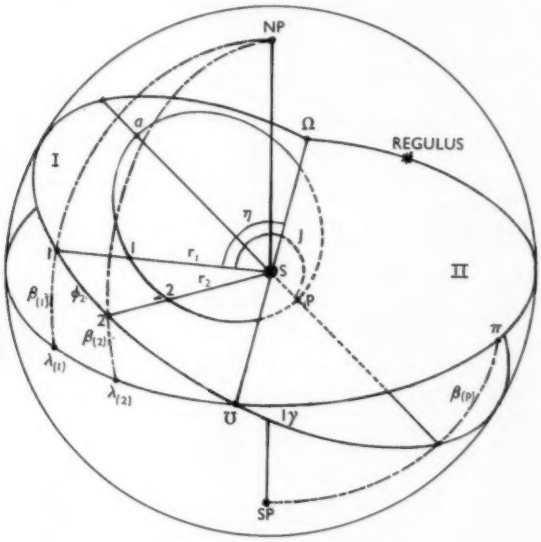


FIG. 8

For position 3 of the spaceship we find in the same manner:

$$\sin \gamma = \frac{\sin \beta(3)}{\sin (\eta + \phi_3)} \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (20)$$

Thus:

$$\frac{\sin \beta(1)}{\sin \eta} = \frac{\sin \beta(3)}{\sin (\eta + \phi_3)} \quad \ldots \quad \ldots \quad \ldots \quad (21)$$

or:

$$\frac{\sin \beta(1)}{\sin \eta} = \frac{\sin \beta(3)}{\sin \eta \,.\, \cos \phi_3 + \cos \eta \,.\, \sin \phi_3} \quad \ldots \quad \ldots \quad \ldots \quad (22)$$

From which:

$$\tan. \eta = \frac{\sin \beta(1) \,.\, \sin \phi_3}{\sin \beta(3) - \sin \beta(1) \,.\, \cos \phi_3} \quad \ldots \quad \ldots \quad \ldots \quad (23)$$

In this manner the position of the ascending node is fixed.

The angle γ between planes I and II is determined by (19):

$$\sin \gamma = \frac{\sin \beta(1)}{\sin \eta}. \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (24)$$

Now we have to convert the co-ordinates of plane I into those of plane II (Fig. 8) and therefore use equations (1) and (2):

Equation (1): $\sin \beta = - \cos \delta \,.\, \sin \alpha \,.\, \sin \epsilon + \sin \delta \,.\, \cos \epsilon.$

Equation (2): $\cos\lambda = \dfrac{\cos\delta \,.\, \cos\alpha}{\cos\beta}$.

In this case:

$\epsilon = \gamma$ $\qquad\qquad$ $\theta = 0$

α is the longitude on plane I, counted from the ascending node, thus:

$\alpha = \eta + \phi$.

Now: $\psi = \theta + \phi$. $\qquad\qquad$ Thus: $\alpha = \eta - \theta + \psi$.

λ is the longitude on plane II, counted from the ascending node, or the *draconic longitude*, designated by $\lambda\Omega$.

Thus equation (1) becomes:

$$\sin\beta = -\sin(\eta - \theta + \psi) \,.\, \sin\gamma \ldots \quad \ldots \quad \ldots \quad (25)$$

Equation (2) becomes:

$$\cos\lambda\Omega = \frac{\cos(\eta - \theta + \psi)}{\cos\beta} \quad \ldots \quad \ldots \quad \ldots \quad (26)$$

Now when λ is the Regulus-longitude of the spaceship and ω is the Regulus-longitude of the ascending node, then:

$$\omega = \lambda - \lambda\Omega \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (27)$$

or: $$\omega = \lambda(1) - \lambda(1)\Omega \quad \ldots \quad \ldots \quad \ldots \quad (28)$$

For position 1: $\psi = \theta$.

Thus, according to (26): $\cos\lambda(1)\Omega = \dfrac{\cos\eta}{\cos\beta(1)}$. $\quad \ldots \quad \ldots \quad \ldots \quad (29)$

For the perihelion: $\psi = 0$. $\qquad\qquad$ Thus: $\alpha = \eta - \theta$.

Therefore the latitude of the perihelion is:

$$\sin\beta(p) = -\sin(\eta - \theta) \,.\, \sin\gamma \quad \ldots \quad \ldots \quad \ldots \quad (30)$$

The draconic longitude of the perihelion is:

$$\cos\lambda(p)\Omega = \frac{\cos(\eta - \theta)}{\cos\beta(p)}. \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (31)$$

The Regulus-longitude of the perhelion is:

$$\pi = \lambda(p)\Omega + \omega \quad \ldots \quad \ldots \quad \ldots \quad \ldots \quad (32)$$

5 (1). *Example.*

Applying equation (23) we find:

$$\tan\eta = \frac{-\sin 8^\circ\,54' \,.\, \sin 31^\circ\,48'}{-\sin 2^\circ\,48' + \sin 8^\circ\,54' \,.\, \cos 31^\circ\,48'} = -0{\cdot}9858.$$

$$\eta = 315^\circ\,25'.$$

Equation (24): $\qquad \sin\gamma = \dfrac{-\sin 8^\circ\,54'}{\sin 315^\circ\,25'} = 0{\cdot}2204.$

$$\gamma = 12^\circ\ 44'.$$

Equation (29): $\cos \lambda(1)\Omega = \dfrac{\cos 315^\circ\ 25'}{\cos 8^\circ\ 54'} = 0{\cdot}7209.$

$$\lambda(1)\Omega = 316^\circ\ 6'.$$

Now: $$\lambda(1) = 272^\circ\ 6'.$$

Thus, according to equation (28):

$$\omega = 272^\circ\ 6' - 316^\circ\ 6' = -\ 44^\circ\ 0' = 316^\circ\ 0'.$$

Equation (30):

$$\sin \beta(p) = -\ \sin (315^\circ\ 25' - 183^\circ\ 15') = -\ 0{\cdot}1633.$$

$$\beta(p) = -\ 9^\circ\ 24'.$$

Equation (31):

$$\cos \lambda(p)\Omega = \frac{\cos (315^\circ\ 25' - 183^\circ\ 15')}{\cos 9^\circ\ 24'} = -\ 0{\cdot}6803.$$

$$\lambda(p)\Omega = 132^\circ\ 52'.$$

Equation (32):

$$\pi = 132^\circ\ 52' - 44^\circ\ 0' = 88^\circ\ 52'.$$

In Fig. 6 the complete orbit of the spaceship is shown projected on the plane of the ecliptic, with the positions of the planets. Besides the observed positions 1, 2 and 3 are shown the positions 4 and 5, when the spaceship reaches the ascending node and the perihelion.

6. The Accuracy of the Observations

All calculations of distances in the solar system are based on the length of the semi-major axis of the orbit of the Earth, the Astronomical Unit. The latest determination of this unit (Spencer Jones, 1941) has given the value of 149,680,000 km. $\pm$ 36,000 km. This means that a determination of the position of a spaceship, at Earth-distance from the Sun, besides the errors of observation, has an uncertainty of 36,000 km., that is about three times the diameter of the Earth.

Now space is roomy and 36,000 km. do not matter so much if we are millions of kilometres from the nearest planet. But on approaching a planet some other means must be used for an exact determination of the distance. Besides, in the vicinity of a planet we have no longer only the Sun's attraction to consider, for the attraction of the planet cannot be neglected and the simple equations of a Keplerian orbit are no longer valid. Therefore, radar, or radio-sounding must be used.

If we put the accuracy of the measuring of the angles on the photographic plates at one second of arc, this will cause an error of about one in a hundred thousand or of about 1,500 km.

If we limit the accuracy to one second of arc we are not troubled by corrections, which would be necessary for greater accuracies. For instance we do not need to bother about the parallaxes of the fixed stars.

α Centauri has the greatest parallax of all of the brighter fixed stars, and this parallax is only 0·76″. Regulus has a parallox of 0·049″. Only when we come

to the very confines of the solar system, to the aphelion of Pluto, will the parallaxes of the stars become perceptible. At this distance the parallax of Regulus is 2·5″ and of α Centauri 40″. But α Centauri lies far from the ecliptic and will seldom be used for orientation.

The error caused by the aberration of light is small, because we have to consider only the difference in the aberration of a star at the centre of the photographic plate and a star at the edge of the plate, only 5° apart. At the normal speeds in free flight in the solar system this error will not exceed a few seconds of arc and can be corrected easily.

TECHNICAL REVIEW

Compiled by E. T. B. SMITH, B.Sc., A.F.R.Ae.S., G.I.Mech.E.

Satellites and rockets in aid of the International Geophysical Year continue to take pride of place in this issue, but we do endeavour to have something for everybody. Readers are invited to suggest fields in which they are interested to help improve the coverage of "Technical Review."

Satellite Instrumentation

H. E. Lagow, of the Rocket Sonde Branch, Naval Research Laboratory, discusses, in a paper presented to the A.R.S. Annual Meeting in 1955 (*J.P.*, June, 1956) the problems involved in the provision of satisfactory satellite instrumentation and in overcoming the difficulties of environment. He makes the following points:—

The orbit plane can be determined by the launcher direction, but the eccentricity depends on the burnout velocity of the final stage: the experimenter should be prepared for the satellite's altitude to vary between 300 and 1,000 km. above the Earth's surface, and upon these altitudes will depend the choice of location of ground telemetry stations. The vehicle's "lifetime" will depend on the air density at perigee, and as this is not known at all accurately, the lifetime will have to be determined from observation.

The shape of the satellite should be spherical, so that drag forces will be independent of satellite orientation. The instrumentation should be capable of operating with the vehicle spinning about an arbitrarily directed axis, and it will be less costly in weight to telemeter the vehicle's attitude than to try and control the attitude in a given manner.

In order to save weight, all electronic gear should be transistorized, on account of the saving on both the instruments and the batteries: until these can be recharged (by solar power, perhaps) they will set the limit to the instrumented lifetime.

Accelerations of zero to perhaps 100 g will have to be catered for, as in contemporary rocket vehicles.

The satellite's temperature will be determined almost entirely by the radiation balance, since convection and internal dissipation of energy will be negligible. The sources of radiation are direct sunlight, reflected sunlight (earthshine and moonshine) and infra-red radiation from the Earth. By selecting a surface material with the proper emissivities in the visible and infra-red

regions, it is theoretically possible to adjust the satellite's temperature. Apparently the night-time temperature, independent of emisivity, will be about − 35° C., while for a polished metal skin (ratio of emissivities in infra-red and in visible light about ½) the day-time equilibrium temperature would be about 115° C. With emissivity ratios of 2 and 1 the temperature would be 30° C. and 70° C., respectively.

The heat capacity of the satellite will prevent these temperatures from being reached, and if the capacity be great enough to make the day/night variation negligible, an emissivity ratio of 2 will give a mean temperature of 10° C. However, the surface emissivities will change due to bombardment with ions, atoms, molecules and meteoric fragments, and there is at present no way of allowing for this: temperature instrumentation will have to be provided in the first instrumented satellite.

Pressures in and around the satellite will present some problems: some experiments will need the residual pressure near the satellite (due to the liberation of gases absorbed on and absorbed in the surface of the vehicle) to be very low, and careful attention will have to be given to the diffusion of gases through the walls: the vapour pressures of the materials of construction will have to be considered, and must be low compared with the ambient "atmospheric" pressure of 10^{-8} to 10^{-11}mm.Hg. The effects of meteoric puncture will have to be considered.

There are forces which will act on the satellite and which will slow it down and ultimately destroy it: these include collisions with air molecules and dust particles, and the effects of escaping gas from the vehicle and energy losses due to eddy currents induced by the Earth's magnetic field. The calculation of the drag forces requires an estimate of density and two assumptions regarding the collision process; the molecules may be reflected either specularly or diffusely, and they may leave the surface with velocities corresponding either to the temperature of the surface or to the incident velocity. Sänger (NACA, TN 1270) indicates that for a normal flat plate the drag coefficient varies by a factor of 2 for the various assumptions at speeds greater than 7 km./sec., but Ashley (*J. Aero. Sci.*, February, 1949) shows that for a sphere the drag coefficient may be between 2·0 and 2·4, while for a particular ogival-nosed cylinder $C_D = 3{\cdot}8$ for diffuse reflection and 0·065 for specular reflection. The drag pressure on a sphere at 300 km. altitude would be about 5×10^{-3} dynes/cm.², which is about 100 times the solar radiation pressure, and at higher altitudes the solar radiation pressure will be greater than the drag. It is estimated that for a vehicle with a cross-sectional loading of 2 gm./cm.² the period of revolution will lengthen by 41 milliseconds per orbit.

Satellite Vehicle Recovery

This problem has two forms of possible solution: the "lift" case, in which the vehicle slows down by coasting in the rarefied atmosphere, and the "drag" case where the vehicle comes in fairly directly and slows rapidly. A proposal which is of the second type is made by R. W. Porter, Chairman of the U.S. Satellite Panel (*AW*, May 14, 1956), in which the payload of the recovery gear is not the entire vehicle but a "data capsule" containing exposed film, and rapid

deceleration is performed by means of a stainless-steel balloon. This balloon should reduce the severe heating and then act as a parachute during descent to Earth. Porter calculates that a ½-lb. package of film can be recovered with a 3 ft. diameter balloon, which would be carried in a folded state in the 3rd stage of Vanguard. As well as the balloon, a retarding rocket would have to be used, and, if the all-up weight of the satellite is to be no more than 21 lb. only 5·5 lb. is left for the actuation receiver, beacon transmitter, balloon, gas, structure and insulation, film, camera and other gear, which is not very much. The retarding rocket is intended to slow the vehicle from 25,000 m.p.h. to 20,000 m.p.h. and make it drop from its orbit; the balloon is then inflated with helium, and, with the film pack, is detached from the gas bottle and retardation rocket. As the balloon falls, at 76 miles it reaches the most critical point; during a period of about 20 seconds a temperature of about 1,200° C. is experienced, with a peak to 1,600° C. If the balloon survives this passage, it then lowers its load so that it drops into the sea at 30 ft./sec. about 20 minutes after leaving its orbit. Although the indicated temperatures are so high, Porter considers that the balloon may stand them, owing to the instability which makes the balloon wobble and so move the stagnation point about on its surface, and to the internal and external coatings of polytetrafluorethylene on the balloon. It has been pointed out (by Dr. Fred Whipple) that meteoritic puncture may defeat the project, and also the fabrication of a stainless-steel balloon may be difficult. It may be possible to use cylindrical shapes rather than spheres with more success.

British Sounding Rocket for the I.G.Y.

Rather more detail than was available in the newspapers is given by an article in *Nature* on April 7, 1956, by F. E. Jones and H. S. W. Massey, which describes the plans of the Gassiot Committee for the I.G.Y. It appears that Treasury support to the extent of £100,000 was agreed on June 29, 1955, to cover equally the design and provisioning of rocket vehicles, by the R.A.E., and the supply of instrumentation equipment, through the Gassiot Committee.

The rocket has been designed to give a suitable compromise between excessive velocity low down (where the drag losses would be prohibitive) and being so slow that there would be undue dispersion of trajectory from the launcher, and the optimum burning time seems to be about 30 seconds. The vehicle is 307 in. long overall, with a body diameter of 17·4 in.; the three fins extend to 44 in. radius. The nose cone is 65 in. long, and the tip (back to 10 in. diameter) can be jettisoned in flight. The all up weight will be 2,500 lb., of which most will be solid propellant: the payload is 150 lb. for a maximum height of 550,000 ft. and with 260 lb. payload a height of 450,000 ft. should be reached.

The launcher is constructed of Bailey Bridge panels, and consists of a triangular-section hollow boom, which can be pivoted from a nearly vertical position, about its centre; the launching angle is variable through $\pm 10°$ in azimuth and from 75° to 95° in elevation, to allow for varying winds. The launcher is 81 ft. high, standing on a three-legged base 72 ft. in diameter, and weighs 35 tons. It can fire Aerobee as well as the Gassiot vehicle.

If the dispersion be excessive, a boost motor may be used to increase the launch velocity: in any case, the initial launches will be made from the wide open spaces of Woomera.

The actual experiments to be carried out with the aid of the Gassiot vehicle are as follows:—

Wind, temperature and pressure by observations of grenades fired at intervals along the trajectory, and mass spectrometry for ion sampling—University College, London. Ionosphere conductivity and relative concentration of electrons to negative ions—University of Birmingham. Relative intensities of spectral lines of molecular oxygen, the altitude of the night air glow, and emissions from sodium and potassium—Queen's University of Belfast. Electron densities—University College, Swansea. Skin temperature and altitude, and sun seeker for measurements of solar radiation and ultra-violet photographs of the sun—R.A.E.

Progress in Propellants

A note in *Jet Propulsion*, June, 1956, gives the following information:—J. P. L. has investigated the storage of red fuming nitric acid in aluminium and 18–8 stainless steels at 130° F. for several months. 0·5 per cent. Hydrogen fluoride passivates the metal surfaces and reduces the corrosion, and the production of excess gas pressure is eliminated by concentrations of 12–16 per cent NO_2 and 2–3½ per cent. water in the acid. Redstone Arsenal, Huntsville, Alabama, have given some data on tetranitromethane. The specific gravity is 1·64, the oxygen content is 69·5 per cent. and the specific impulse is 224 seconds with a hydrocarbon fuel. The freezing point is 57° F. and the compound is liable to detonate.

Solid propellants in the news are those made by Aerojet, who are looking at systems of synthetic linear polymers as binders for inorganic crystalline oxidisers; by Atlantic Research, of Alexandria, Va., who are using plastisols, polyurethanes and epoxies; by the Naval Powder Factory at Indian Head, Md., in cast solids, synthetic materials, and nitroguanidine; by Phillips Petroleum at McGregor, Texas, in ammonium nitrate, carbon black, and synthetic rubber; and by Thiokol Chemical Corpn., Trenton, N.J., in perchlorates and polysulphides.

Hypersonic Research

In *Jet Propulsion*, July, 1956, A. Hertzberg, of the Cornell Astronautical Laboratory, Buffalo, N.Y., has described the use and function of hypersonic shock tubes at the Laboratory. He states that the shock tube is one of the few laboratory instruments which permits the generation and use of those gas temperatures which will be encountered in hypersonic flight (such as re-entry of a vehicle into the atmosphere): at $M = 12$, the stagnation temperature is about 4,000° K., at which the air has ionized to such an extent that its electrical conductivity is comparable with that of sea water. Shock tubes with parallel working sections are limited, on the flow Mach number achieved, to about $M = 3$ for real air. However, the use of a divergent nozzle permits the flow to accelerate, and flow Mach numbers of up to $M = 5·2$ or more can be achieved.

The tunnels may be driven by combustion of a mixture of oxygen and

hydrogen, diluted with helium, and since the testing time is of the order of a few milliseconds, the observations are made largely by schlieren photography with a high-speed movie camera. The author also describes the use of thin-film resistance thermometers for estimating the heat transfer into nose cones; these are about 1 micron thick, and respond accurately at rates of several thousand °C./sec. Work is also in hand on the chemical aspects of hypersonic flow, such as the formation of nitric oxide.

Some further notes in *Aviation Week*, July 2, 1956, indicate that Mach numbers of about 20 have been reached in a helium blow-down tunnel at Princeton and a shock tunnel at Cornell, while a molecular beam tunnel at the University of California produces a Mach number of 40 (at a pressure of 10^{-8} atmosphere). Free flight systems include the Lockheed X-17 hypersonic three-stage solid propellant rocket vehicles, and various guns which use gases of low molecular weight as propellants: Ames Aeronautical Laboratory (NACA) has such a gun which uses helium, and which can fire a $\frac{1}{4}$-in. diameter projectile at up to about 15,000 ft./sec.

Other methods of achieving high velocities include the use of an electric arc to heat gases and of an electric coil which would burst when a large current is applied; the projectile would be designed as part of the coil, but neither of these methods has yet been exploited.

Life Enclosures

In *Jet Propulsion*, July, 1956, Simons and Parks of Holloman Air Development Centre, describe the design of capsules used for taking small animals (mice, hamsters, guinea-pigs and monkeys) for balloon rides up to 100,000 ft. for 24 hours or so. The capsule is formed of two hemispherical aluminium spinnings of 24 in. diameter, and possesses simple and ingenious devices for control of temperature, carbon dioxide content, oxygen content, and humidity. It is of interest that the environmental control was adequate at all flight conditions, but the temperature and humidity rapidly became intolerable, as soon as the capsule came to earth (presumably in the New Mexico desert).

Aero Digest, April, 1956, notes that workers at Holloman and Wright Air Development Centres are interested in putting up an animal-carrying satellite, but that it is unlikely that this will be done until the Vanguard firings are successful—the re-entry/landing problem will have to be solved for this project.

The Colours of Jupiter

In a letter to the Editor of the *Journal of Chemistry and Physics*, in the June, 1956 issue, F. O. Rice, of the Catholic University of America, in Washington, D.C., discusses some possible mechanisms for the production of the great variety of colours, steady and evanescent, which are observed on the planet Jupiter. The atmosphere consists of methane mixed with ammonia and perhaps hydrogen and helium, and the average temperature of the observable part of the upper atmosphere is about − 150° C., so that it is difficult to understand the origin of the colours or of their continual changes.

The author points out that one possibility is that the colour changes may be due to meteorological activity which whirls small crystallites of material, some

being coloured, into the upper atmosphere. The bands of Jupiter move at different angular rates, and this, together with the relatively rapid rates of heating and cooling caused by the 10-hour period of rotation, is likely to give rise to storms of very great magnitude.

A second possibility is that certain coloured chemical compounds are formed in Jupiter's atmosphere. The blues and greys may be due to sodium dissolved in liquid ammonia, the shade depending on the temperature. The great Red Spot may be due to the formation of cuprene, a polymer of acetylene, which forms when methane is illuminated by light of very short wavelength.

The author then describes another approach which he and his co-workers have investigated: they have produced a number of "reactive species" which have such properties, and can exist in such surroundings, that they may be expected to be formed in the atmosphere of Jupiter. These "reactive species" are described below:—

Empirical formula	Colour	Transition temperature, °C.	Remarks
$(NH)_n$	Blue	− 125	Changes to ammonium azide, NH_4N_3.
$(NH_2NH)_n$	Yellow	− 178	May be tetrazane, $NH_2NHNHNH_2$: gives HN_3 at transition temperature.
$(CH_3N)_n$	Colourless	− 195	Thermal decomposition of methyl azide, CH_3N_3.
$(CH_3CH_2)_2N$	Light yellow	− 160	Electric discharge on $(C_2H_5)_2NH$.
$(CH_3)_2N$	Deep green	− 170	From $(CH_3)_2N-N=N-N(CH_3)_2$ in electric discharge.
CH_3S	Yellow	− 170 to − 160	From CH_3S-SCH_3.
$(CH_3)_3CS$	Red	− 170 to − 140	From $(CH_3)_3CS—S(CH_3)_3$.
S_2	Purple	—	Changes from purple to yellow in a few seconds at room temperature and in a few hours at − 80° C.

The presence of ammonia and methane, which yield radicals on illumination with short wavelength ultra-violet light, the low temperature necessary to preserve these species, and the long ages of solar illumination all support the hypothesis, and the electric storms would promote the radical formation.

The workers are measuring the absorption spectra, in the visible and near ultra-violet, of the above species, with the object of comparing their spectra with that of reflected light from Jupiter.

Gravitational Research

Aero Digest for March, 1956, discusses the present state of research on gravity, and mentions the work done by the Gravity Research Foundation in New Boston, N.H., the Institute for Advanced Studies in Princeton, N.J., and other academic places, as well as by such aircraft manufacturers as Convair, Bell, Sikorsky, Lear and Martin. It is believed that gravity has now passed from the purely observable state to where a certain amount of experimentation

can be done, in the field of high energy nuclear and sub-nuclear particles, "which may prove to be basic gravitational energy that is being converted continually and automatically in an expanding universe directly into the most useful nuclear and electro-magnetic forms."

Rockoons and Solar Flares

Aviation Week, July 23, 1956, notes that, in order to aid the study of Solar Flares, solid propellant rockets (Deacon) are being fired from balloons floating at altitudes of some 80,000 ft. Since the phenomena associated with Solar Flares grow and die quickly, the time taken for a rocket to ascend from sea level is prohibitive, and, in the system chosen, the Deacon, with 20 lb. of intruments, can be at an altitude of 70 miles in 90–120 second from leaving the plastic balloon. Telemetry is used to transmit the findings of photon counters, working in the bands corresponding to the Lyman-α line of hydrogen, X-rays, and soft gamma rays.

Weather Forecasting from Meteor Showers

Satellite vehicles have been advocated as a help for weather forecasting, and it is usually assumed that this will be done by transmitting observations or photographs of the growth and movement of cloud systems over the Earth. But they could be used in another way if we accept the new ideas put forward by Dr. E. G. Bowen, of the Division of Radiophysics, Commonwealth Scientific and Industrial Research Organization, Sydney, Australia, in the June 16 issue of *Nature* (**177**, 1121, 1956).

Dr. Bowen points out that forecasters who work from conventional weather charts have had remarkably poor success in telling the man in the street what he really wants to know, namely, whether it is going to rain to-morrow. Most rain, outside the Tropics, is believed to be inititated by ice crystals in those parts of a cloud which have risen so high as to become cooled down below freezing point. But the water droplets which make up the cloud do not automatically freeze when they become supercooled. Ice can only form if it has a suitable nucleus on which to crystallize, and, says Dr. Bowen, "it is well established that the number and properties of the freezing nuclei present in the atmosphere determine whether rain will form in a supercooled cloud." The trouble is that, although the forecasters can predict pretty well whether there are going to be any supercooled clouds, they cannot predict the quantity of "freezing nuclei" of the kind which will initiate rain by becoming centres of growth for ice particles. Freezing nuclei in this context are defined as "particles on which ice crystals can grow from the vapour phase, or else materials which in solution in a water droplet or in contact with it will induce it to freeze."

Dr. Bowen has now made measurements from aircraft of the concentration of freezing nuclei, and has found, rather unexpectedly, that though the measurements "are consistent among themselves in one area or over a few hours at a time, changes of several hundred to one are possible from one day to the next." What is equally surprising, it is found that sudden changes in the concentration of freezing nuclei occur at almost identical times in different parts of the world. For instance, during January, 1955, measurements were made simultaneously

on a mountain peak in Hawaii and from aircraft over New South Wales by Dr. Bowen and over Arizona by three of his colleagues (Smith, Kassander and Twomey, *Nature*, **177**, 82, Jan. 14, 1956).

Averaging these and similar observations in the Januaries of 1954 to 1956, Dr. Bowen shows that there are marked peaks of concentration on the 13th, 22nd and 30th of the month. He then gives long-period averages of January rainfall in the British Isles, United States, South Africa and New Zealand which, again, show peaks at just about these dates.

What possible cause can there be for this striking agreement on or around the same dates each year? Dr. Bowen can suggest only one; meteor showers. Now, it has been shown that the smaller particles of meteoric dust, which have not enough energy to be consumed on entering the atmosphere, take 30 to 40 days to sink from the top of the atmosphere into the region where rain clouds form (Z. Svestka, *Bull. Astro. Inst.* Czech., **2**, 41, 1951). There are three well-known meteor showers: the Geminids of December 13, the Ursids of December 22, and the Quadrantids of January 3; the intervals between these and the peaks of rainfall and freezing nuclei are just about right. Again, the agreement is striking.

Dr. Bowen admits that his hypothesis (he does not yet dignify it by calling it a theory) has aspects which are difficult to accept; for instance, he has not discovered why meteoric dust should be such an active nucleating agent. But, if he is right, there must be enormous short-period changes in the quantity of meteoric dust through which a satellite vehicle would pass, and if by some means the amount could be measured and communicated to Earth, the information could well be of immense value, for it would give the forecasters a whole month's warning.

Experimental Sealed Cabin

Dr. Hubertus Strughold writes in the *Journal of Aviation Medicine* (**27**, 50, February, 1956) of the construction of an experimental sealed cabin for the United States Air Force School of Aviation Medicine. He points out that, at heights above 80,000 ft., the conventional pressurized cabin must be replaced by one that is hermetically sealed, "in which the breathing air is not pumped in from the air outside the craft, but must be taken along at the start. The oxygen consumed by the occupants of the sealed cabin must be replaced from stores within the ship. The carbon dioxide must be removed, and temperature, humidity and odours must be controlled." The experimentation programme, Dr. Strughold writes, "involves two main problems which warrant investigation:

"1. To what extent, and in what direction, are the various climatic factors changed by the presence of occupants in the cabin.

"2. How can these changes be counteracted by physical, technical, or biological means.

"This experimental chamber also can serve as a training device to indoctrinate the occupant with the problems encountered in a closed ecological system and to familiarize him with the procedures necessary to handle any emergency situations such as the failure of the automatic control systems and an eventual

puncture of the cabin itself." He adds that this sealed cabin "is a prototype of the cabin that may be built into future space ships."

But there are two odd features about this article, which is only a couple of pages long. Firstly, although the cabin was delivered by the Guardite Company of Chicago in the summer of 1954, there is no indication that it has yet been used. Secondly, an accompanying photograph shows it against a background of furniture, from which one can estimate that it is no more than two or three feet high—not even big enough to serve as a coffin.

Perspective

To give an idea of the way things have speeded up lately, a quote from B. C. Monesmith, Vice-President of Lockheeds, may be of interest (*AW*, March 26, 1956). He said "This plane (the F-104) could be cruising along minding its own business and be fired at by a 16-in. Naval gun. If the pilot should happen to see it (the shell)—which he could—he could avoid it, swing alongside and inspect it, and then accelerate away from it going in the same direction."

FOR SALE, 4-in. Astro./Terr. Refracting Telescope, by Thos. Cook & Sons. Alt. Azimuth Mounting Sight, telescopic steadying rods, tripod, and several eyepieces, including Sun diagonal. In wooden case, £50.—J. GARRETT, 4, Church Road, Penarth, Glam.

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OBSERVATION OF ARTIFICIAL SATELLITES

By J. Humphries, B.Sc.(Eng.), A.M.I.Mech.E., A.F.R.Ae.S.

In common with most other technical and scientific developments, the liquid-propellant rocket was pioneered by amateurs. Unfortunately the cost of rocket development, even on a small scale, is far higher than in most other fields of scientific endeavour and the work was taken over by professionals at a very early stage. If it had not been for the rocket's obvious military applications the high cost of development would probably have forced the liquid-propellant rocket to remain a curiosity rather than the powerful and reliable source of power that it is today. The amateurs' interest in the rocket was as a means of propulsion in space—the professionals' as a weapon. In consequence the amateur rocket enthusiast has remained, although in very few instances has he carried out any useful practical work. Nevertheless, the theoretical studies carried out by the various interplanetary societies have been serious ones and have had not a little influence in convincing professional and official authorities of the feasibility of space-flight.

It has been the contention of many that the amateur is unlikely to be able to make any useful practical contribution towards the achievement of space-flight and it is therefore very exciting to hear that amateur effort is to be recruited to aid the American artificial satellite programme. The particular phase in which amateurs are to be used is that of optical observation both in the initial phases and at the end of the satellite's life. "Project Vanguard" has been adequately described both in the "Technical Review" section of the *Journal* and in the first issue of *Spaceflight* and so only a few of the relevant details will be repeated here. The satellite itself is to be a 20-in. diameter sphere of highly-polished magnesium, thickness 0·030 in. The total weight, including instruments, is to be about 20 lb. The satellite will be placed into its orbit by a three-stage rocket vehicle. Because of the various uncertainties in the performance and control of the rocket motor and the guidance of the vehicle, the exact orbit of the satellite cannot be prophesied with very great accuracy. It is hoped to establish an elliptical orbit with a minimum perigee of 200 miles and maximum apogee of 800 miles. With a mean altitude of about 500 miles the satellite's period will be about 100 minutes.

The motion of the satellite through the observer's sky will be rapid in comparison to the speed of all other celestial bodies, except meteors. A satellite will cross the entire United States in only 10 minutes and on its closest approach to the Earth will appear, to an observer directly beneath it, to move at about 1·3° of arc per second—it has been said that viewing it will be equivalent to viewing a golf ball travelling at the speed of sound at 40,000 ft. ! None of the satellites now planned will be a conspicuous object. The average apparent visual magnitude under favourable circumstances will be approximately 7, depending upon the exact size, albedo (reflecting power) and distance. When at perigee and directly overhead the satellite may be a magnitude brighter, and at apogee or at a low meridian passage, as faint as 9th or 10th magnitude.

The launching site will be at Patrick Air Force Base near Cape Canaveral in Florida. The firing must be directed over water where shipping has been suspended so that the heavy motors of the first and second stages will not endanger human life and property when they fall. The inclination of the orbit to the equator will be between 30° and 40° so that at the furthest point north the satellite may fly over central Spain. Although the horizon from the satellite will be at a very considerable distance, none of these early satellites will in practice be visible from the British Isles.

Observation of the satellites will be a major operation, both visual and radio tracking will be used. The early satellites will carry a small radio transmitter of 10–50 milliwatts output operating on 108 mc. and weighing, with sufficient batteries to last for several weeks' intermittent operation, only 2–3 lb. The "Minitrack" as this system is known will suffice to establish the initial orbital parameters and to transmit a certain limited amount of information from the satellite's instruments. For really accurate orbital data twelve Schmidt cameras, stationed around the world, are to be used. But before these can be brought into operation the initial orbit has to be determined. This, of course, will be done with Minitrack, but electronic equipment can always fail—usually at the critical moment—and for this reason a visual tracking system largely employing amateurs has been initiated as an insurance policy. "Moonwatch" as the project is called is being organized by the Smithsonian Astrophysical Observatory under the direction of Prof. F. L. Whipple. A *Bulletin for Visual Observers of Satellites* is being published and four numbers have already appeared each dealing with various facets of the project, in particular the organization of the observing teams and the equipment to be used.

Although full details have not yet been worked out, the organization of Moonwatch will be along the following lines. A central computing bureau for receiving and evaluating optical observations will be established, probably at Harvard Observatory. The observing groups in any given geographical area will be informed when an observable satellite is due in their region and if they observe the satellite will send in their observations by radio.

Each qualified group will be under the direction of an experienced leader and each observer must have a pair of binoculars or a monocular of wide aperture and field supported on a mounting that can be directed to a chosen field on his meridian. The recommended basic specifications for a suitable instrument are: objective diameter 45–55 mm, magnification, 6x–7x, field of view, 10–12°. Twenty to thirty observers will be required per team, the instruments being set up with overlapping fields to form an "optical fence" along the meridian. A short-wave radio to give accurate time signals is almost indispensable and a tape-recorder is a useful adjunct to make a permanent recording of any sighting.

During the observing period each observer will be conversant with the fainter stars as they pass across his field of view and will thus be able to report the precise path of a satellite across the field should he be fortunate enough to see it. On seeing a satellite the discoverer will signal the timekeeper and recorder of the group to prepare them for accurate time determination. The group leader and his assistant will check the authenticity of the sighting and if they are satisfied other observers will then follow the satellite across the sky and an

accurate time record as it occults various stars will be made. At the earliest possible moment the observations will be transmitted to the central computing bureau and the computing machine will immediately incorporate the observations into an improved solution for the motion of the satellite.

It is obvious from the above short description that a high degree of organization will be involved in the amateur programme, that experienced observers are needed and that results from individuals are of no use. Indeed the positions of the observing sites must be accurately known and each team is to be graded according to its reliability. It seems probable that the majority of these teams will be organized by Universities and such bodies as astronomical societies. At the Seventh International Astronautical Congress held in Rome this year, a committee for co-operation with the I.G.Y. authority on this matter was formed. It recommended that, in order to get things started, the individual astronautical societies should contact their national I.G.Y. representatives to discuss possible action in their own countries. To this end the B.I.S. has formed a committee to investigate the possibility of the B.I.S. taking part in observing the satellites. Although we shall not be able to participate in the first few launchings, later orbits will almost certainly bring the satellites over Great Britain.

The Committee held its first meeting recently and decided that before much detailed planning could be undertaken it was necessary to know the probable number of members who would be interested and able to take part in the programme. Up to 30 people will be needed for each watching post and the Committee is not therefore thinking at the moment in terms of more than one post. The qualifications for potential observers will be an appreciable experience of astronomical observing and a willingness to turn up to a certain number of practices and, of course, to turn out when suitable satellites have been launched. Would potential observers please write to J. Humphries, 97 Churchill Avenue, Southcourt, Aylesbury, Bucks., giving brief particulars of experience, including type of instruments used. Although it is almost certain that any activity will have to be confined to the London area, replies from all parts of the British Isles will be welcome in order to ascertain the degree of interest in the project.

In some of the satellites the Minitrack equipment may be left on throughout the complete orbit. Although the Committee has no further information at the moment on this proposal, it would also like to hear from members interested in helping in a Minitrack receiving station. The number of people involved would be much smaller and only those professionally engaged in electronics or radio, or amateurs with very extensive experience, should apply. Please send details of qualifications and experience.

NOTES AND NEWS

Yugoslav Astronautical Society

The membership of the Yugoslav Astronautical Society has doubled over the past year and now stands at 92. Last year the programme included 22 lectures in Belgrade and other cities and in April last the Society co-operated with Radio

Belgrade in a Children's Hour programme on a trip to the Moon which was broadcast in eight languages.

Norwegian Astronautical Society

Last year the "Norsk Interplanetarisk Selskap" changed its name into "Norsk Astronautisk Forening." It now numbers 40 members.

Rocket and Space-flight Exhibition

The Deutsches Raketen-und Raumfahrtmuseum organized a big exhibition on rockets and space-flight in Gotebörg, Sweden, in August this year, to be followed later by a similar exhibition in Stuttgart, Germany.

The Museum also co-operated with the U.S. Embassy in Western Germany in staging a special exhibition on the theme of high-altitude research with rockets, balloons and radio under the title of "Space Unlimited," during the International Industrial Exhibition in Berlin, September 15 to October. Over 10,000 visitors came to the Exhibition daily which was so successful that special arrangements were made for the display to be reopened again from October 20 to November 4. On display were many original electronic apparatus, e.g. a small transmitter for the Vanguard satellite, rocket motors from Reaction Motors, Inc., Bell X-I "Skyrocket," combustion chambers, and full-size models of the Bumper and three-stage Vanguard rocket projects. It is understood that the exhibition will be put on at other German cities.

Course on Orbital and Satellite Vehicles

A two-week intensive course on "Orbital and Satellite Vehicles" was offered by the Massachusetts Institute of Technology, during their summer session. Six hours of lectures were given daily from August 6–17 on such topics as the astronomical background to space-travel, rocket design fnd performance, new possibilities in propulsion, instrumentation of unmanned vehicles and the physiological problems of man in space.

Tuition for the course was $200 (£70).

International Congress on Rockets and Guided Missiles

The *Association pour l'encouragement a la Recherche Aeronautique* is arranging an International Congress on Rockets and Guided Missiles to be held in Paris, December 3–8, 1956.

In addition to a number of lectures, a public exhibition of rockets and guided missiles and associated equipment will be staged and visits to factories and laboratories will be arranged. In addition, it is planned to include engine and rocket tests and a Banquet will conclude the proceedings. A special programme has been arranged for ladies who accompany the Congressionists.

The address of A.E.R.A. is 1, Rue de Courty, Paris, 7e, France.

OFFERS invited for unbound *Journals*, Indexes to some, in good order. *B.I.S.*, Vol. 6, Nos. 4–7; Vols. 7–13 complete; Vol. 14 except No. 4; Vol. 15, 1–4 inclusive. *A.R.S.*, October, 1947 to March, 1951, except 79 and 80: Vol. 23, 1–3, 6; Vol. 24, 1–5; Vol. 25, 2, 3, 6, 9/1 and 2, 10–12 inclusive. *P.R.S.*, Winter, 1947 to Fall, 1949 and 2 Bulletins (total of 8).—Apply TREGURTHA, Tremar, Underhill Road, Stoke, Plymouth, Devon.

Changes in Publication Dates of the Journal

As from January 1, 1957, the *Journal* will become the quarterly technical publication of the Society, and will be published as follows:—

(*a*) Jan.-Mar. issue on or about February 1.
(*b*) Apr.-June ,, ,, May 1.
(*c*) July-Sep. ,, ,, August 1.
(*d*) Oct.-Dec. ,, ,, November 1.

Spaceflight is also being issued quarterly and issued to members free of charge, i.e. a total of eight publications will be issued next year instead of six as previously.

Obituary

Professor A. M. Low

We deeply regret to record the death of Professor A. M. Low in September last. The Professor was a President of our Society during the immediate pre-war years, at a time when very few men of science cared to be associated with the idea of spaceflight, and was well-known for his engaging sense of humour and deep interest in almost every branch of science. He was responsible for the first guided missile, which was flown over Salisbury Plain in 1916–17: it was launched from a ramp by compressed air and designed to be radio and/or gyro controlled. Built by the R.F.C. under conditions of great secrecy, it was intended for anti-aircraft attack and is now in the Imperial War Museum. In 1917 the Professor also designed a conductor-wire controlled rocket.

At one time Assistant Professor of Physics at the Royal Ordnance College, the Professor engaged in a life of great activity and is best-known by his multitude of popular books on science. It is a matter of profound regret that he did not live to see the establishment of the first artificial satellites and the achievements of a science which he supported so strongly against much personal abuse and ridicule in the immediate pre-war years.

Mr. Charles Bein

Mr. Bein, a staunch supporter of the Society from its very earliest days, passed away in September last after a painful illness which he bore with fortitude.

In the early days of 1935, Mr. Bein co-operated enthusiastically in the projects for Zucker's rocket post experiments near Brighton and in the Hebrides, and during his life he assembled one of the world's most important collections on the history and development of rocket airmail experiments during the period 1928–38.

Mr. Tom Bourdillon

We regret to record the death of Mr. Tom Bourdillion on August 5, while climbing in the Alps. Many members will recall Mr. Bourdillon's recent lecture on the use of portable breathing apparatus, during which he showed a number of photographs taken on the Mount Everest Expedition, of which he was a member.

ABSTRACTS

Edited by J. HUMPHRIES.

A list of abbreviations of titles of journals was included with the 1954 index and addenda have been published in subsequent issues of the *Journal*. A further addendum is given below.

Bol. Asoc. Arg. Interplanet.	Boletin de la Asociacion Argentina Interplanetaria.
J. Acoust. Soc. Amer.	Journal of the Acoustical Society of America.
J. Washington Acad. Sci.	Journal of the Washington Academy of Sciences.

ASTRONAUTICS

See also abs. nos. 325 and 345.

(308) **Trip to the Moon: a Russian view.** K. Stanyukovitch. *Aviation Wk.*, **61** (9), 36–8 (*Aug.* 30, 1954). Translation of an article from the Russian publication *News*, dealing with the problems in a general manner.

(309) **"Mouse," a minimum unmanned artificial satellite for astronomical investigations.** S. F. Singer. *Bol. Asoc. Arg. Interplanet.*, **8**, 2–12 (*April*, 1955) (*in Spanish*). A translation by P. M. Viarnes of the paper presented by Dr. Singer in the Third Symposium on Space Travel, held in the Hayden Planetarium in May, 1954.

(310) **First steps of Astronautics: Unmanned Vehicles.** T. M. Tabanera. *Bol. Asoc. Arg. Interplanet.*, **8**, 24–9 (*April*, 1955) (*in Spanish*). An account of the application of crewless instrument-carrying rocket vehicles.

(311) **Orbit lifetimes of U.S. artificial satellites.** H. B. Ketchum. *J. Space Flight*, **7** (8) 1–5 (*Oct.*, 1955). Initial investigation of 110 lb. 20 in. diameter satellites shows that lifetime will vary from 15 minutes at 125 km. altitude to almost a year at 500 km. (4 *refs.*).

(312) **The influence of spaceflight on engineering and science.** M. W. Rosen. *J. Washington Acad. Sci.*, **46**, 79–84 (1956). Historical review of the development of rockets and the idea of space-flight.

(313) **Flight mechanics of the satelloid.** K. A. Ehricke. *Aero. Dig.*, **73** (1), 46–8, 50, 52, 54 (*July*, 1956). Satelloids are low-thrust rocket-powered vehicles capable of long-duration sustained flight in the extremely rarefied regions of a planet's atmosphere. Their operational characteristics are discussed. (1 *ref.*)

(314) **Factors affecting the lifetime of Earth satellites.** N. V. Petersen. *Aero. Dig.*, **73** (1), 74–6, 78, 80, 82 (*July*, 1956). The various types of air drag in a diffuse gas are discussed and the results applied to examples of satellites. Curves are given showing the lifetimes of various satellites. (4 *refs.*)

(315) **More Vanguard details are revealed.** *Aviation Wk.*, **65** (9), 31 (*Aug.* 27, 1956). Details of the satellite itself. This is to be a 20-in. sphere of highly-polished magnesium, 0·030-in. thick.

(316) **First Vanguard tests scheduled this fall.** D. A. Anderton. *Aviation Wk.*, **65** (13), 26–8 (*Sept.* 24, 1956). Review of technical papers delivered to the Seventh International Astronautical Congress.

(317) **Relativity and space-travel.** H. Dingle and W. H. McCrea. *Nature*, **178**, 680–2 (*Sept.* 29, 1956). Further correspondence on the time dilatation effect in flight at speeds near that of light. (4 *refs.*)

(318) **Towards space-flight.** *Engng.*, **182**, 478 (*Oct.* 12, 1956). Review of the Seventh International Astronautical Congress.

(319) **Seventh International Astronautical Congress.** T. Nonweiler. *Nature*, **178**, 832–3 (*Oct.* 20, 1956). Review of some of the technical papers presented.

MATERIALS

(320) **Improvements relating to rocket motor and like structures.** Carborundum Company. *Brit. Pat. No.* 749,231 (*May* 23, 1956). Various silicon carbide mixtures for use as rocket nozzle liners.

PROJECTILES

(321) **Range measuring systems using pulse techniques.** P. Kernan. *Aero Dig.*, **73** (1), 32–4, 38, 40, 42, 44 (*July*, 1956). Reviews methods of measuring velocity and distance. The selection of the best system depends on whether it will be manual or automatic, high precision or lightweight and whether the target will be fast or slow.

(322) **Guidance—the brains of missiles.** D. T. Sigley, N. I. Korman and G. Worsley. *Aero Dig.*, **73** (1), 22–31 (*July*, 1956). Review of types of guidance, including beam riding, homing, command, inertial and stellar. Errors in guidance are also discussed as are the specific problems relating to satellite guidance. (**4** *refs.*)

(323) **Optimum rocket trajectories.** B. D. Fried and J. M. Richardson. *J. Appl. Phys.*, **27**, 955–61 (*Aug.*, 1956). It is shown that under certain simplifying assumptions maximum range is achieved if the thrust attitude angle is held constant during the entire powered flight. This is exact for constant gravitational field and a good approximation in an inverse square law field provided that the distance covered during the powered flight is small compared with the radius of the Earth. (**2** *refs.*)

RADIO AND ELECTRONICS

(324) **Transistors telemeter small missiles.** C. M. Kortman. *Electronics*, **29** (7), 145–7 (*July*, 1956). A germanium photocell used with a three transistor f.m./f.m. telemeter counts revolutions of a 2-in. diameter missile in flight. Other channels measure acceleration and pressure.

(325) **Space navigation challenges engineers.** P. A. Castruccio. *Aviation Wk.*, **65** (10), 82–4, 87, 89, 91, 94, 97, 99 (*Sept.* 3, 1956). Considers radar methods of navigation and communication within the next twenty years. Using a broad beam of 24° a range of 20×10^6 miles should be possible for an input power of 1 megawatt. Ship-to-ship communication would also be possible under some circumstances using narrow beams and scanning systems for making contact. Radar techniques for detecting and avoiding meteors are considered impractical.

ROCKET MOTORS

See also abs. nos. 320 and 343.

(326) **The Walter process, a process for obtaining impulse energy.** E. Kruska. *Z. Ver. dtsch. Ing. (V.D.I.)*, **97**. Pt. 1, 65–70 (*Jan.* 21, 1955); Pt. 2, 271–7 (*March* 21, 1955); Pt. 3, 709–13 (*July* 21, 1955); Pt. 4, 823–9 (*Aug.* 21, 1955) (*in German*). Part 1. The Walter process is based on the decomposition of high concentration hydrogen peroxide (80 per cent.) which is highly exothermic. Thermodynamic properties of decomposition products and of combustion with various fuels, including hydrazine hydrate. Pt. 2. The design of combustion chambers, including that for the 109–509 aircraft motor and various experimental chambers utilizing a pre-combustion chamber for non-self-igniting propellants. Schemes for expulsion of propellants—pressurized and turbo-pump. Pt. 3. The application to submarines and torpedoes—production of gas to drive turbines. Pt. 4. Layouts of actual units, 109–500 "cold" assisted take-off unit, 109–502 "hot" assisted take-off unit, 109–507 glide-bomb unit, R. II 203 and 109–509 series of main aircraft propulsion units and V-1 catapult. (**1** *ref.*)

(327) **A study of vibrations originating in the fuel lines of a rocket-motor.** J. A. Landoni. *Bol. Asoc. Arg. Interplanet.*, **8**, 13–23 (*April*, 1955) (*in Spanish*). A discussion of the conditions for the generation of oscillations in pipeline systems, and their effect on rocket combustion stability.

(328) **Theoretical study of low frequency instabilities in liquid propellant rockets.** M. Barrère and J. J. Bernard. *O.N.E.R.A. Publ. No.* 79, 41 pp. (*July*, 1955) (*in French*). The instability in a combustion chamber with a constant rate of injection is considered in relation to the design parameters of the chamber and the ignition delay of the propellant mixture. The influence of the supply system on the fluctuations of the rate of injection are then examined for mono- and bi-propellants. The method of determining the critical frequencies for complex conditions is expounded. (**14** *refs.*)

(329) **Rocket propulsion in helicopters.** L. M. Slawecki. *J. Space Flight*, **7** (7), 5–7. (*Sept.*, 1955). Short descriptions of existing units. (**4** *refs.*)

(330) **Flames possessing supersonic velocity produced by the burning in air of rocket exhaust gases.** H. Behrens and R. Rössler. *Z. angew. Phys.*, **8**, 237–44 (1956) (*in German*). Measurements made on solid propellant rockets of the temperatures reached in secondary combustion.

(331) **Survey of organ-pipe oscillations in combustion systems.** A. A. Putnam and W. R. Dennis. *J. Acoust. Soc. Amer.*, **28**, 246–59 (*March*, 1956). Mainly a review of 53 references on the subject. The systems discussed range from simple singing-flame burners to industrial furnaces and liquid-propellant rocket motors.

(332) **Improvements in or relating to propellant supply systems for jet reaction motors.** Minister of Supply. *Brit. Pat. No.* 749,050 (*May* 16, 1956). A method of controlling the starting flow in a pressurized liquid propellant rocket motor so that it does not become excessive.

(333) **Improvements in or relating to jet reaction motors.** Minister of Supply. *Brit. Pat. No.* 749,670 (*May* 30, 1956). Injector for liquid propellant rocket motor using transverse injection.

(334) **Armstrong Siddeley Screamer.** *Flight*, **70**, 160–4 (*July* 27, 1956). Full details of the unit together with a history of its development. This is an aircraft motor of 8,000 lb. thrust using liquid oxygen, kerosine and water as propellants.

(335) **Improvements in or relating to jet propulsion engines combined with a rocket.** Societé Nationale d'Etude et de Construction de Moteurs d'Aviation. *Brit. Pat. No.* 754,141 (*Aug.* 1, 1956). Scheme for placing a rocket combustion chamber centrally in a turbo-jet or ram-jet nozzle.

(336) **Improvements in thermal nuclear reactors, in particular for aircraft propulsion.** H. P. G. A. R. von Zborowski. *Brit. Pat. No.* 754,559 (*Aug.* 8, 1956). Includes schemes for working fluid nuclear rocket.

(337) **Napier spurs British rocket operations.** *Aviation Wk.*, **65** (13), 59 (*Sept.* 24, 1956). Data on and photos of several Napier rockets—the Scorpion, N.R.E.11 and N.R.E.17.

(338) **Combustion is focus of rocket meeting.** R. Cushman. *Aviation Wk.*, **65** (14), 32–3 (*Oct.* 1, 1956). Review of papers presented at the A.R.S. Fall Meeting.

ROCKET PROPELLANTS

See also abs. nos. 326, 338 and 343.

(339) **Combustion problems in solid propellant rockets.** T. Marklund. *Artill. Tidskr.*, **85**, 167–90 (1955) (*in Swedish*). A summing up of existing theories and problems in internal ballistics and experimental techniques for determination of the burning rate with special reference to the Research Institute of National Defence, Sweden.

(340) **Improvements in or relating to catalyst stones for the decomposition of concentrated hydrogen peroxide.** Minister of Supply. *Brit. Pat. No.* 749,431 (May 23, 1956).

(341) **Characteristics and application of rocket propellants.** K. R. Stehling. *Aero. Dig.*, **73** (1), 56–8, 60 (*July*, 1956). Discusses existing propellants and their properties. (4 *refs.*)

(342) **Liquid hydrogen as a rocket propellant.** A. Pastuhov and C. L. Jewett. *Aero Dig.*, **73** (3), 35–9 (*Sept.*, 1956). Considers problems of manufacture, storage and transfer.

ROCKET PROPULSION

See also abs. no. 323.

(343) **Some problems in rocket drives.** O. Lutz. *Deutsches Forschungsanstalt Luftfahrt Bericht F.B.S.* 11/54, 17 pp. (*April*, 1954) (*in German*). Deals with the principles of the Mélot nozzle for thrust augumentation, using the atmospheric air, calculation of reaction enthalpies for high-temperature combustion, rate of reaction related to the deviation from equilibrium conditions, the use of nuclear energy and the development of self-igniting propellants.

(344) **Mach 2 rocket sled.** *Engng*, **181**, 546–7 (*June* 29, 1956). The Convair sled is propelled by 12 solid-propellant rockets on a 10,000 ft. long track—water brakes are used for stopping it. The sled is being used to investigate the effect of rain on supersonic missiles.

(345) **Flight mechanics of photon rockets.** E. Sänger. *Aero Dig.*, **73** (1), 68–70, 72–3 (*July*, 1956). The use of photon rockets for interstellar flight and a consideration of time dilatation effects.

REVIEW

Guide to Mars

(By P. Moore. F. Muller Ltd., London. 124 pp. 10s. 6d. 1956.)

Interest in the planet Mars, always high, is increasing as the time for its comparatively close approach to Earth draws near. This book has been written to provide a non-technical summary of knowledge and ideas about the planet.

On occasion the author experiences difficulty in trying to explain science in a non-technical way. For instance, the comparison on page 17 between Ptolemy's epicycles and the motions of the planets probably could not be understood by anyone who did not already know what it was about. The same trouble occurs again on pages 27 and 29 in references to the effect on weight, and density of the atmosphere, of the reduced pull of gravity at the surface of Mars as compared to the Earth. A statement of Newton's Law of Gravitation and consideration of the effects of both mass and radius would have been as simple as the description given, and certainly more accurate and informative.

On pages 37 and 38 the author discusses whether the dark band round the pole cap is real or an optical illusion, but makes no reference to the dark rim which is clearly visible on the photograph facing page 32.

The statement on page 30 that by the year 1840 it was known that there was vegetation on Mars would not be accepted by the many astronomers who have investigated the problem since then, nor will many be happy with the remark on page 48 that the dark areas on Mars are as permanent as the continents of Earth. The suggestion on the same page that scientists who have put forward non-vegetative theories to try to explain the dark areas did so because they disliked admitting life on planets other than the Earth is unjust.

The author's belief that no practical selenographers of today have much doubt that the moon is a volcanic waste (page 55) makes one wonder where practice ends and theory begins.

On page 81, the remark that any political leaders who declare war should be put in criminal lunatic asylums is a strange inclusion in a book about Mars the planet, while the suggestion, page 104, that scientists are not allowed to concentrate upon true progress instead of upon barbarous militarism will be news to everyone.

The map of Mars would have benefited from being drawn a little beyond longitude 180° on each side, because all but one of the features which reach 180° east do not reappear at 180° west.

Although the book has many faults, the greater part of it is interesting and informative, and will be appreciated by all those who develop an interest in Mars from news of its coming close approach to the Earth.

V. C. REDDISH.

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In view of the fact that a List of Members can be issued only at infrequent intervals, the Council has decided to provide information on the elections made at each meeting, as it is felt that this is likely to prove of interest to other members.

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LAWRENCE HALFORD YOUNG, B.Sc., Water Development Dept., P.O. Box 312, Blantyre, Nyasaland.

The following elections were made at the Council Meeting of October 6, 1956:—

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JOHN EDGELEY, 4, Cleveland Square, W.2.
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JOHN ANTHONY REGAN, 106, Lower Tritonville Road, Sandymount, Dublin, Eire.
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WILLIAM DEVINE, 47, Eglesfield Road, South Shields, Co. Durham.
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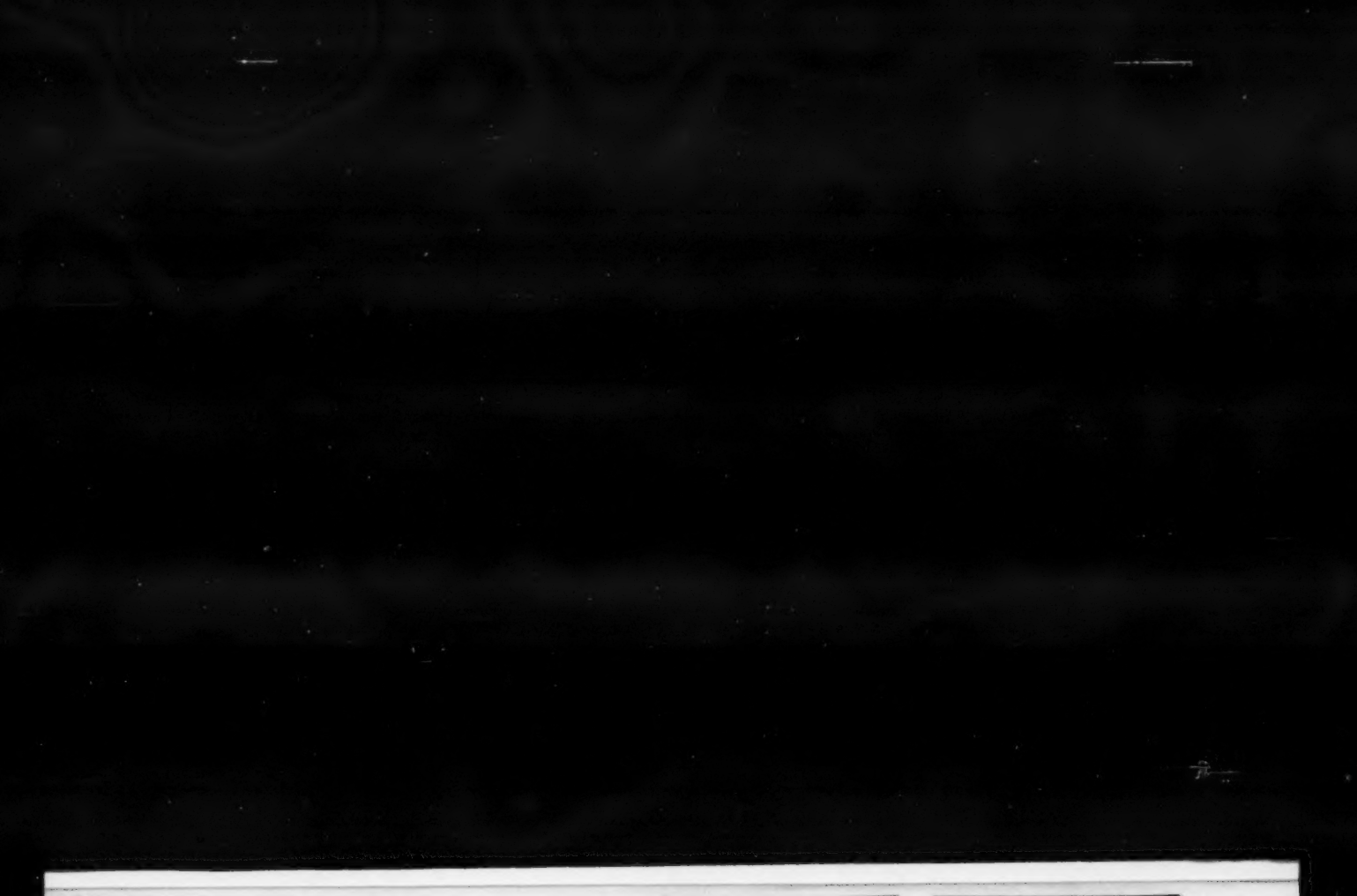
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